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UNITED STATES NAVY

PROJECT SQUID

FIELD SURVEY REPORT

LIQUID PROPELLANT ROCKETS

Volume II, Part 2

30 June 1947

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Field Survey Report

Volume II, Part 2

by

WILLIAM C. HOUSE

Princeton University

Princeton, New Jersey

30 June 1947

[redacted]

Princeton University, the central management organization of Project SQUID, arranged for the preparation of the *Field Survey Report*, under Contract Number N6ori-105, Task Order III, with the Office of Naval Research, Navy Department.

This report was prepared by the Technical Survey Group of Project SQUID as a cooperative effort of Princeton University and Engineering Research Associates, Inc. Engineering Research Associates was given primary responsibility for the preparation of these reports in accordance with the provisions of Task Order II under Purchase Order Number 08451 with Princeton University.

TECHNICAL SURVEY GROUP

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FOREWORD

The *Field Survey Report* on liquid propellant rockets and pulse jet engines was prepared at the suggestion of the Policy Committee, in order that the fundamental research in Project SQUID might be related to other projects and programs of research in this field, and to problems arising in the development of rocket and pulse jet engine equipment.

In order to fulfill this purpose the *Field Survey Report* had to be more than a brief outline of the work of each contractor, but time did not permit it to be prepared as a monograph in each branch of the field of propulsion. The choice of presentation of the work in each volume of the report was governed in part by the amount of available information and by its relation to the rese. rch now being sponsored by Project SQUID.

The Policy Committee will use the *Field Survey Report* as a basis for adjustments in the research program of Project SQUID, in order to ensure a more effective attack on the fundamental problems in the field of propulsion. The Policy Committee hopes that this report may also be useful to scientists conducting research and development in fields relating to propulsion, and to members of government organizations responsible for the planning and integration of research programs in propulsion.

HUGH S. TAYLOR, Chairman
Policy Committee, Project SQUID

PREFACE

The Field Survey Report was prepared by the Technical Survey Group, Project SQUID, under the direction of Engineering Research Associates, Inc.

The assembly of the material and the preparation of each part of the report was undertaken as a group effort, to which the staffs of both Princeton University and Engineering Research Associates, Inc., have contributed. Mr. F. A. Parker, Project Organizer, and Mr. W. C. House, Chief Technical Aide, of the central administrative staff of Project SQUID at Princeton served as members of the Technical Survey Group and prepared Volume II. In addition, Professor J. V. Charyk of the Aeronautical Engineering Department at Princeton visited the California Institute of Technology and furnished basic information concerning the research program there. He also offered many helpful suggestions with regard to several parts of Volume I.

In the preparation of this report the members of the Technical Survey Group have received the assistance, counsel and cooperation of representatives of the War and Navy Departments and other Government agencies, and of representatives of academic and industrial laboratories who are under contract to the government for research and development in this field.

The authors are indebted to a number of scientists who have reviewed each part of the report and have offered much constructive criticism. The authors also wish to express their appreciation for the assistance which was so generously given by representatives of the Office of Naval Research and of the Bureau of Aeronautics.

THE TECHNICAL SURVEY GROUP

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I. SUMMARY

This report considers problems confronting workers in the field of liquid rocket development and summarizes the work throughout the nation being directed toward the immediate solution of those problems. It contains a brief review of the historical background in the United States, the underlying principles, the design difficulties, and the usefulness of liquid rocket propellant power plants. The long-range studies are discussed in other reports of this Survey (see page ii).

The technical effort of eighteen known agencies designing, building, or testing rocket equipment or components of rocket systems is discussed. These agencies employ approximately 567 people of engineering caliber and 1,367 technicians, mechanics, etc., on 127 individual task assignments under Army, Navy, or other Federal contract. The information presented was obtained through visits to the agencies concerned and conversations with their personnel.

Specific recommendations are made for a coordinated program of future work.

Literature references are listed at the end of this report.

II. CONCLUSIONS AND RECOMMENDATIONS

When reduced to its fundamentals an organized overall rocket development program must be directed towards the following three points:

1. Reduce specific propellant consumption,
2. Reduce weight of all structural parts,
3. Increase utility and reliability.

To proceed with the greatest efficiency towards these goals, the following program is recommended. It is based on the considered opinions of the experts consulted in the compilation of this report.

A. General Program

1. A theoretical review by an unbiased agency of all of the known and proposed liquid propellants. This would include the establishment of a practical and consistent means of comparison of systems to the end that a selection of a small number (ten or less) of combinations upon which all national effort could be directed. The selection should be based on these four fields; aircraft assisted takeoff, aircraft superperformance, missile launching, missile propulsion.

2. The establishment of a few well-equipped rocket test stations throughout the country rather than a large number of poorly or modestly equipped stations.

3. The preparation of suitable textbooks in the field of jet propulsion on a national scale as a means of organizing the tremendous mass of data now in existence. This has a secondary but no less important object in providing a better means of thoroughly and accurately educating newcomers to the field.

B. Detailed Program

A detailed program should consist of the following:

1. The study of the chemical kinetics of combustion and the rate of reaction of the various superior propellant combinations with the purpose of increasing the speed of combustion; thus perhaps making possible a reduction in chamber size with a consequent saving in weight. This should include the study of combustion catalysers of all conceivable types such as fuel additives and catalytic walls of the chamber, etc. This should also include the study of the effect of flow patterns on combustion with the view of establishing optimum combustion chamber shape.

2. The continued gathering of thermodynamic data on present, new or proposed propellants at high chamber pressures and temperatures.

3. The theoretical and experimental study of the fundamental problems and parameters of liquid-liquid streams mixing in the high pressure and high temperature conditions of the rocket combustion chambers.

4. The theoretical and experimental study of the fundamental problems and parameters of heat transfer from the hot gases in the combustion chamber to the motor walls, through the walls, and then into a gaseous or liquid coolant. This shall take into account the details in the respective boundary layers and

should be extended to cover conditions encountered in sweat or porous wall cooling and in ceramic lined walls.

5. The continued development of porous ceramic and metal wall liners for liquid rocket motors.

6. The study and development of mechanical construction methods to produce a large number of exactly controlled passages through rocket motor walls with the intent of producing optimum practical cooling conditions. This may be considered as a hybrid between multiple-hole film cooling and porous wall sweat cooling.

7. The continued study of ceramic liners for rocket motors. Particular emphasis should be placed on the liners which are naturally formed in the operation of diborane motors.

8. The development of a rational and practical method of detailed stress analysis in rocket motors based on experimentally determined pressures and loads under running conditions. This should include a study of rocket motor construction methods and possible improvements from the standpoint of ease of fabrication, light weight, and optimum combustion chamber shape.

9. An expansion of the present program in ducted rockets or rocket-ramjets in both the theoretical and experimental directions in order to improve the performance by establishing the parameters affecting the mixing of hot gas streams at high velocity with lower velocity cold streams.

10. Analysis of the problems of pressure-fed rocket power plants with the view of possible weight and volume reduction through heating of the pressurizing gas.

11. Analysis of the problems of combustion chamber pressure in a pumpfed rocket engine in order to determine the optimum pressure for maximum overall specific impulse.

12. The study of the problems of pumps and pump drives from mechanical, hydraulic, and thermodynamic standpoints for the purpose of showing favorable lines of attack in the design of light efficient propellant *pump systems* for rocket engines.

13. The study of the problems of rotating seals for high-speed liquid oxygen and acid pumps.

14. Analysis of the heat transfer conditions on turbine blades for the exhaust jet-driven turbopump. It should provide specific answers to the following questions:

a. How long can a blade remain in the jet without damage and upon what factors does this depend?

b. What is the effect of adding additional motors around the periphery of the turbine? What is the limiting number?

c. What is the mechanism of blade cooling?

d. What will be the effect of altitude on blade cooling?

e. Can the air cooling be replaced by propellant vapor? If so, how should such coolant be applied to the blades?

15. The development of instrumentation for the rapid measurement of flame temperature and jet velocities. Methods should be developed for hot gas sampling to aid in chemical kinetic and thermodynamic studies of combustion.

III. INTRODUCTION

A general discussion is presented below of the history, the fields of usefulness, the problems involved, and the factors influencing the development of liquid rocket propulsion systems. As many as possible of the known problems are enumerated as an outline for the evaluation of the present national engineering research and development program.

A. Brief History of Liquid Propellant Rockets in the United States

Dr. Robert H. Goddard ran the first liquid propellant rocket motor on November 1, 1923. This rocket was mounted on a vertical test stand and utilized liquid oxygen and gasoline as propellant. It was the first known use of liquid propellants in the United States and apparently in the world. Dr. Goddard began some original theoretical work in 1912 and 1913 at Princeton University on solid propellant rockets and actual experimental investigations began in 1915 at Clark University. Here a successful demonstration was made of a rocket operating in a vacuum, thus clearly illustrating one major advantage of the rocket. A report by Dr. Goddard, entitled "A Method of Reaching Extreme Altitudes," was published in 1919 by the Smithsonian Institution (18). This report covers his early work on solid propellant rockets. On March 16, 1926, a liquid rocket system of Dr. Goddard's made a short but successful flight. On May 31, 1935, at Roswell, New Mexico, a liquid oxygen gasoline turbopump-fed rocket, designed by Goddard, ascended to an altitude of 7,500 ft. The motor developed approximately 800-lb. thrust, and since it was fed by a self-sufficient pumping system it can be considered truly remarkable, for it was almost completely the work of one man and in terms of the life of the liquid rocket field it was well ahead of its time. This work was sponsored by the Daniel and Florence Guggenheim Foundation.

In March 1930 the American Rocket Society was founded, and two years later a group of members ran their first tests. Numerous tests were run thereafter, with varying success. In December 1941 four members of the above group formed a company, Reaction Motors, Incorporated, on the strength of the U. S. Navy Bureau of Aeronautics' interest in jet assisted takeoff. On March 23, 1942, this group delivered to the Navy its first regeneratively-cooled rocket motor. The motor developed 100 to 110-lb. thrust utilizing liquid oxygen and gasoline as propellants, and was capable of being throttled from 50% to 100% rated thrust. The duration was indefinite and depended on the supply of propellants. A subsequent improved model of this motor developed 130 lbs. thrust and was incorporated in the Naval Engineering Experiment Station CML-1N in a power plant described in Section IV of this report.

The Guggenheim Aeronautical Laboratory, California Institute of Technology (GALCIT), began sponsoring some solid propellant rocket research in 1936, but it was not until 1939 that active work began in the field of liquid rockets under Army Air Forces sponsorship. This ultimately led to the flight-testing of an A-20 airplane, using two 25-second uncooled 1,000-lb. thrust red fuming nitric acid-aniline rocket units for jet assisted takeoff. The flight tests began on April 7, 1942. In July 1942 a 500-lb. thrust uncooled red fuming nitric acid-aniline GALCIT unit was delivered to the Navy, and in September 1942 a similar 1,000-lb. unit was accepted.

Since the GALCIT group was part of a non-profit organization and not set up for production work, the GALCIT group formed a corporation in February of 1942, known as Super Power, Inc., for the production of any items requested by the services. Later the name of this organization was changed to Aerojet Engineering Corporation. The two GALCIT units described above were built by this organization.

Robert C. Truax, while still a midshipman at the United States Naval Academy, ran a gaseous oxygen-gasoline rocket motor in September 1937. In 1939 he successfully demonstrated a jet velocity of over 5,000 ft./sec. His continued interest and effort led to the establishment of a Bureau of Aeronautics project at the U. S. Naval Engineering Experiment Station (USNEES), Annapolis, Maryland, for the development of a suitable jet assisted takeoff device for flying boats. This project was established in May 1941. Subsequent development led to successful flight tests of a PBY aircraft in May, 1943 with two 35-second 1,500-lb. thrust red fuming nitric acid commercial aniline (DU-1) rocket units (Fig. 1). These units were later adapted to mixed acid and monoethylamine. In July 1942 Dr. Goddard joined the USNEES group and continued his development work under Bureau of Aeronautics, Navy Department sponsorship.

Figure 1. U. S. Naval Engineering Experimental Station DU-1 jet assisted takeoff unit.

The subsequent developments of the above-mentioned five agencies are shown in Table III of Section IV of this report. This represents the total of the nation's effort in a period when rocket research people were generally looked upon as dreamers. Since the conclusion of the war and the establishment of the usefulness of the liquid rocket, the group of agencies in the field has increased fourfold.

B. The Fields of Usefulness of the Liquid Rocket

The two principal advantages of rockets over all other known forms of propulsion are its ability to function in a vacuum and its independence of thrust with respect to velocity. The consequence of these facts are that the rocket increases its thrust or efficiency with altitude, and that in a vacuum it will continue to accelerate as long as the propellants are supplied, or until it approaches the speed of light. It is the only known means of propulsion for travel in outer space. Other less obvious advantages are summarized as below:

1. It will function under water.
2. It will develop a greater thrust per unit weight than all other known means of propulsion at speeds under 100 mph for periods up to approximately 60 seconds.
3. It will develop a greater thrust per unit volume than all other known means of propulsion at speeds under 100 mph for periods up to approximately 300 seconds.
4. It will develop full thrust in less than a second after starting the motor.

The advantages cited above make the rocket useful, and in many cases mandatory, for propulsion of the following:

1. Guides missiles and sounding rockets.
2. Interceptor aircraft.
3. Superperformance of aircraft.
4. Jet assisted takeoff for missiles and aircraft.
5. Torpedoes and underwater missiles.

Additional factors which must be considered in the choice of a power plant for a given application are:

1. Storage and handling of propellants (hazards, etc.).
2. Reliability.
3. Volume from the standpoint of installation and stowage on board ship.
4. Availability and cost of propellants.
5. Complexity of design and cost of manufacture in comparison with other systems.
6. Changes in efficiency of present designs by design improvements and propellant improvements.
7. Improvement in efficiency or usefulness of competitive propulsion systems.
8. Life, from the standpoint of repeated operation.

C. Principles and Design of Liquid Propellant Rocket Motors

A liquid propellant rocket motor consists of three essential parts: the injector, the combustion chamber, and the nozzle (Fig. 2). The injector provides a means of entrance for the combustible fluid or fluids to the combustion chamber where it is burned at a high pressure, and the nozzle then provides a means of

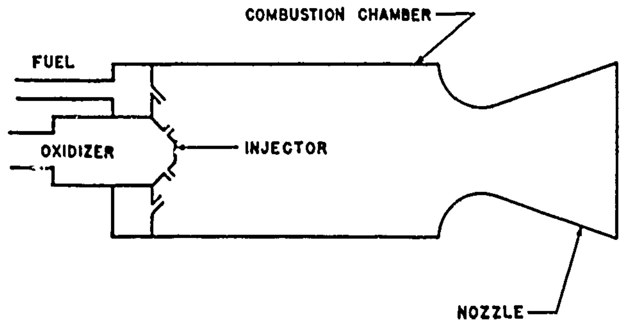

Figure 2. Components of a liquid propellant rocket motor.

converting the high pressure gases into a high velocity jet from which a thrust results.

The relation of the thrust to this effective jet velocity is given by the following equation:

$$F = mc \tag{1}$$

where F = thrust (lbs.)
m = mass flow rate (lbs./g.sec.)
c = effective jet velocity (ft./sec.)
g = gravitational constant (ft./sec./sec.)

The thrust and mass flow rate are the basic measurements of performance taken on rocket test stands. From these the effective jet velocity is calculated by the above equation.

Another basic parameter of the rocket motor is known as the *characteristic velocity*. It can be calculated from fundamental data by the following equation:[1]

$$c^* = \sqrt{\frac{\gamma R T_c / M}{\Gamma'}} \tag{2}$$

where: c^* = characteristic exhaust velocity
γ = ratio of specific heats of exhaust gases
R = universal gas constant
T_c = absolute temperature of gases
M = average molecular weight of gases

$$\Gamma' = \gamma \left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma + 1}{2(\gamma - 1)}}$$

The values of T_c, M, and Γ' are calculated from the thermochemical data for the specific propellant combination used and the parameter c^* may be considered as a measure of merit of the propellant. It is seen from equation (2) that a high combustion temperature and low molecular weight of exhaust products is conducive to large values of c^* and consequently large values for the effective exhaust velocity.

[1]See Reference 21, page 146.

Experimentally c* is determined from the following equation:

$$c^* = \frac{P_c\, f_t}{m} \qquad (3)$$

where: P_c = absolute chamber pressure (p.s.i.a.)
f_t = area of nozzle (sq.in.)
m = mass flow rate (lbs./g.sec.)

In practice it is usually found that the experimental c* varies from the theoretical value by approximately 10%, which is probably due to incomplete combustion, internal losses, etc.

Combining equations one and three we arrive at a new equation:

$$F = \frac{c}{c^*} P_c\, f_t \qquad (4)$$

Replacing c/c* with C_F we get:

$$F = C_F\, P_c\, f_t \qquad (5)$$

The parameter C_F is known as the *thrust coefficient.* It is related to the characteristic velocity c* and the effective exhaust velocity. It is best to determine C_F by test, but where no test data are available, its theoretical value can be obtained from thermochemical data by using the following equation:[2]

$$C_{F_{th}} = \Gamma' \sqrt{\frac{2}{\gamma-1}\,\eta_t} + \left(\frac{P_e-P_o}{P_c}\right)\frac{f_e}{f_t} \qquad (6)$$

where: $C_{F_{th}}$ = theoretical thrust coefficient

$$\eta_t = 1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}$$

P_e = absolute pressure in nozzle exit
P_c = absolute chamber pressure
P_o = absolute outside pressure
f_e = nozzle exit area
f_t = nozzle throat area

The above equation shows the factors that affect the thrust coefficient. Figure 3 shows the effect graphically for $\gamma = 1.2$. From this it is apparent that nozzle design and thrust coefficient are materially affected by chamber pressure and outside pressure.

Two additional parameters are commonly used in establishing the performance of a rocket motor. They

[2]See Reference 21, page 143.

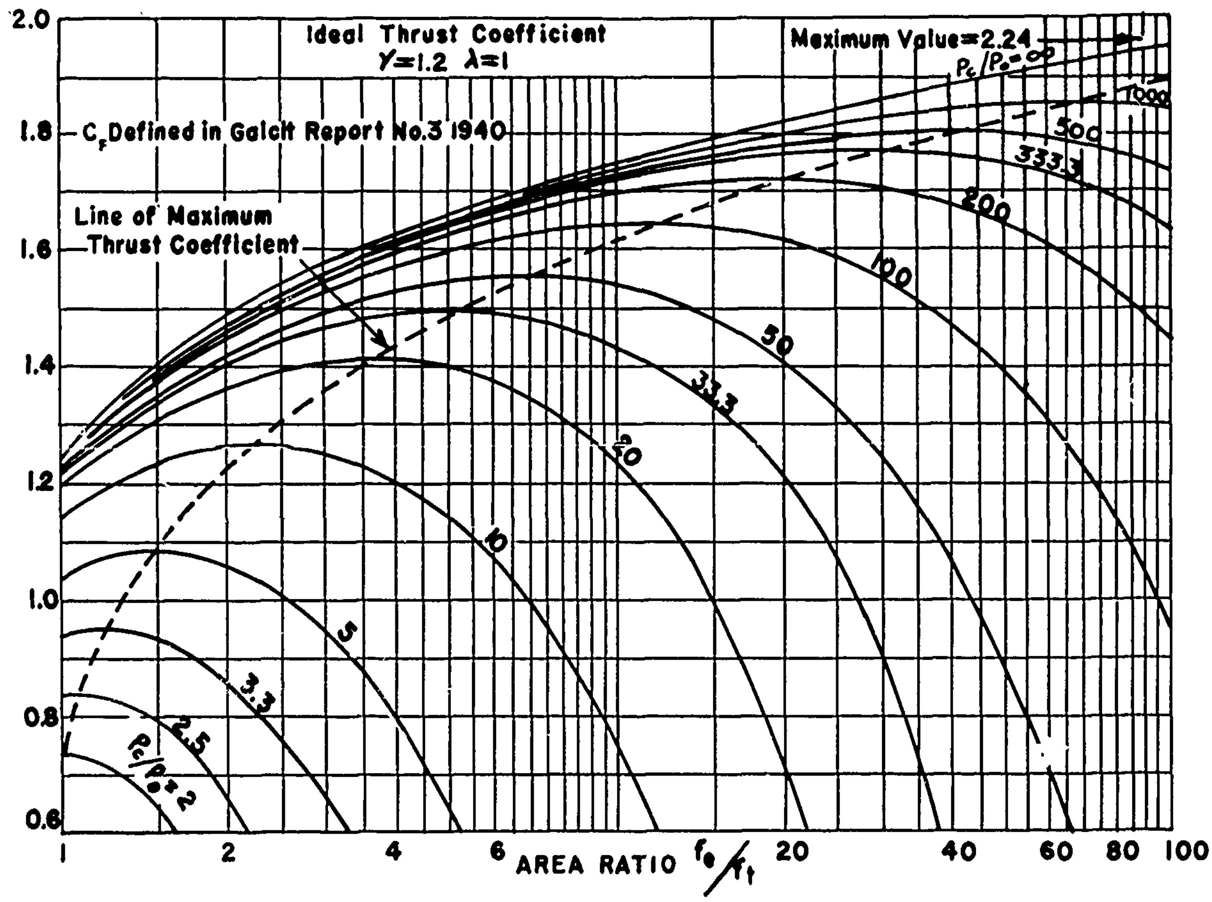

Figure 3. Variation of C_F for various nozzle throat-exit ratios.

are directly related, since they are a measure of propellant consumption, and one is simply the inverse of the other.

Specific propellant consumption, W_{sp}, is determined by:

$$W_{sp} = \frac{mg}{F} = \frac{w}{F} = \frac{g}{c} = \frac{g}{c^*C_F}$$

Specific impulse, I_{sp}, is the inverse of W_{sp} and may be written:

$$I_{sp} = \frac{Ft}{W} = \frac{F}{w} = \frac{c}{g} = \frac{c^*C_F}{g}$$

where: t = duration (sec.)

W = total flow (lbs.)

$w = \frac{W}{t}$

It appears impossible to establish a performance parameter based on rocket motor combustion chamber geometry. It is reasonable to assume that the combustion volume must be large enough to ensure complete combustion. Immediately several variables come to mind which will affect the time of combustion, such as type and location of injector, completeness of mixing, kinetics of the reaction of propellant combination, and path of particles from injector to nozzle throat. Practical design and construction considerations have led to the almost universal use of the cylindrical combustion chamber with the propellant injection system at one end and the nozzle at the other. Because of the lack of data on the effect of motor geometry, it cannot be concluded, however, that the aforementioned arrangement is the optimum. For very large motors of 20,000-lb. thrust and greater, the cylindrical combustion chamber may no longer be practical. For example, the combustion chamber of the A-4 motor is very largely the entrance to the nozzle.

The parameter most commonly used to define motor geometry is L^*, the ratio of the combustion chamber volume to the area of the nozzle throat. Thus a minimum L^* will produce the lightest motor for a specified geometrical arrangement. That this parameter is useful may be shown experimentally by tests with varying L^*. The characteristic velocity will increase up to a certain L^* for a given propellant and will then remain substantially constant for larger values. An equation may be derived for the optimum condition, when the propellants are assumed completely mixed at the injector and the combustion is complete at the end of the combustion chamber. But since this equation requires knowledge of the reaction kinetics for the propellants, it is of little practical value, because of the difficulty of determining reaction time, mixing time, etc. For these reasons optimum L^* is determined experimentally. The magnitude of the parameter L^* as defined above does not have general applicability. It is a variable that depends on detail chamber geometry, scale effect, chamber pressure, injector design, and propellant combination.

Certain agencies have adopted the length of the combustion chamber as a parameter. For moderate changes in thrust, other conditions being similar, this has a useful value. However, for a very large increase in thrust, a simple sketch (Fig. 4) will show the difficulty which will be encountered.

The ratio of the diameter of the combustion chamber to the diameter of the nozzle throat is also significant, since it determines to a large extent the velocity of the gases in the chamber and hence the heat transfer to the walls.

A summary of the factors that affect the detail design of the motor are listed below:

1. thrust required.
2. propellant.
3. structural materials and permissible weight.
4. cooling system
5. injection.
6. ignition.
7. life and operation.
8. operation altitude (nozzle design).
9. fabrication.

A brief discussion of these variables follows below:

1. Thrust Required. The thrust required determines the physical dimensions of the motor and its parts. The lower limit appears to be of the order of 50 to 100 lbs., owing to practical minimum size of injector orifices, etc. From present design considerations, there appears to be no upper limit to the size of motor which may be built.

Scale does not appear to have any pronounced effect on c^*, although there is some indication of a reduced C_F for motors under 500-lb. thrust. This, however, may not be substantiated by further accurate testing.

It is important to note the effect of chamber pressure on c^*. There is an increase in jet velocity with increasing chamber pressure, and on this basis the motor would become smaller for increased chamber pressure, as can be seen from equation 5. The decreased area of throat would then decrease the chamber volume if L^* were to remain constant.

2. Propellants. The physical properties of the propellants affect design in various manners. These may be summarized as follows:

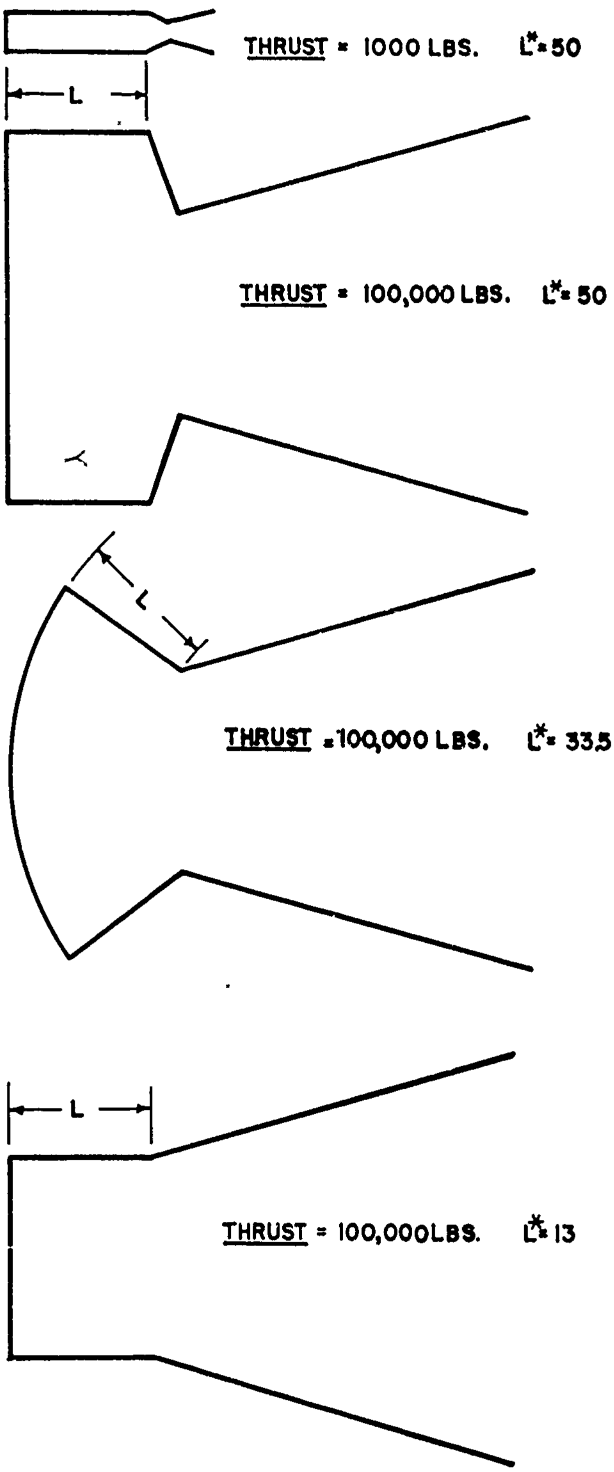

Figure 4. Geometrical scale effect for constant chamber length.

a. heat capacity as a coolant.

b. viscosity effects on:

1. injector orifice coefficients,
2. pressure drop in cooling jacket,
3. mixing qualities.

c. combustion:

1. temperature.
2. kinetics.

Chamber pressure and combustion volume or L* are directly related to kinetics and temperature of combustion. The choice of materials of construction is also dependent on propellants from the standpoint of corrosiveness and stability.

3. Structural Materials and Permissible Weight. Once the materials have passed the requirements suited to the propellants (see above), the strength of the material at operating temperature determines the wall thicknesses required for the chosen chamber pressure and feed pressure in the case of regenerative motors. The theoretical static pressures on the inner walls in conjunction with the cooling jacket pressure for regenerative motors can be used for determining these wall thicknesses, but no tests to date have been run to check variation from the theoretical values. This, of course, will have a pronounced effect on the heat transfer, as well as the load distribution, particularly in the nozzle. Design for repeated operation or one-shot operation will determine the weight to some extent by minimum gauge material selected, as will be shown later.

4. Cooling System. The magnitude of the cooling problem can best be envisioned by a typical example of the quantities of heat involved. In a 1,000-lb. thrust motor the heat liberated is approximately 9,850 Btu per second.[3] The problem is not as severe as one might imagine from first examination. Only two to three per cent of the total heat is involved in the cooling cycle of a regeneratively cooled motor. Approximately 40% of the heat is converted into kinetic energy of the jet and 60% remains as heat energy in the exhaust gases.

Figure 5 is a curve showing the relative decrease in cooling surface required on a geometrical basis for increasing thrust rocket motors with a constant L*.

Basically there are four types of cooling systems: heat capacity, regenerative, film, and transpiration or sweat cooling. Heat capacity type motors cannot be strictly said to have a cooling system, since no new medium is brought into play to continue cooling. However they are included here for unity of presentation.

a. Heat Capacity type motors are simply those in which the walls are thick enough to absorb the heat transferred to them during the running period without melting (Fig. 6). These are usually called uncooled motors.

In general these motors are heavy and can only be used for periods of twenty-five seconds or less; but in

[3]See Reference 21, page 394.

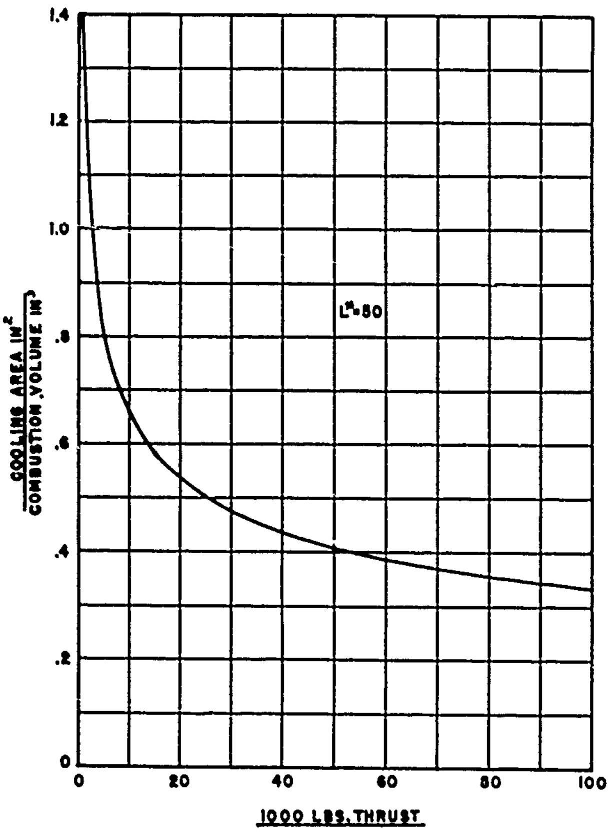

Figure 5. Relative cooling area decrease for constant L* motors.

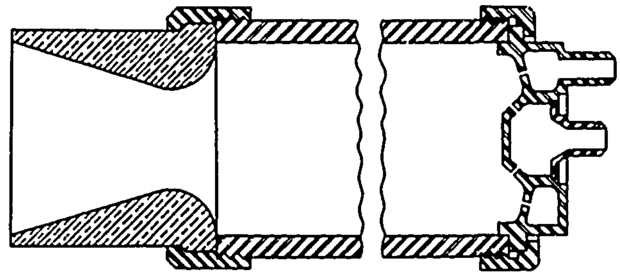

Figure 6. Uncooled rocket motor.

the case of nitromethane or hydrogen peroxide cold units this period is much longer, due to the lower chamber temperatures involved. Ceramic coatings would probably increase the life if they could be prevented from cracking off due to thermal effects. As mentioned above, the weight of these chambers makes them useful only in assisted takeoff of aircraft applications, where weight requirements are not so severe. The specific fuel consumption is also usually higher (since these motors are normally run with an excess of fuel) in bipropellant systems, to maintain a lower chamber temperature.

b. Regeneratively cooled motors are the most commonly used today. Figure 7 indicates their general nature.

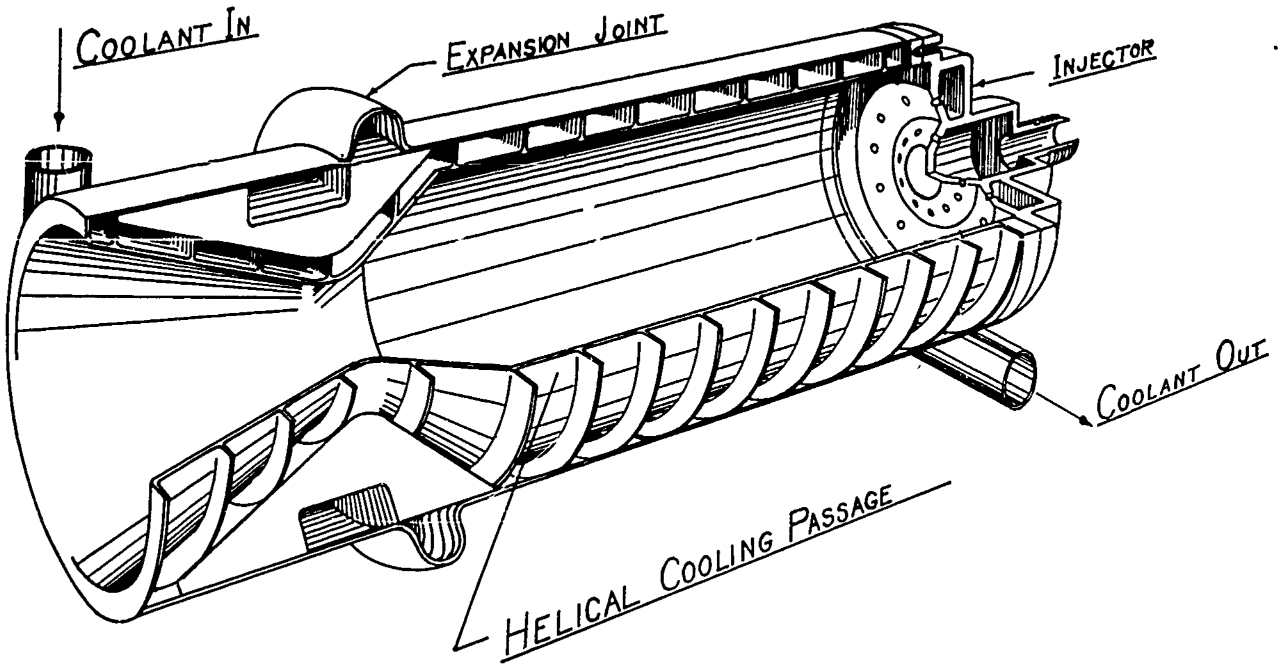

Figure 7. Regeneratively cooled rocket motor.

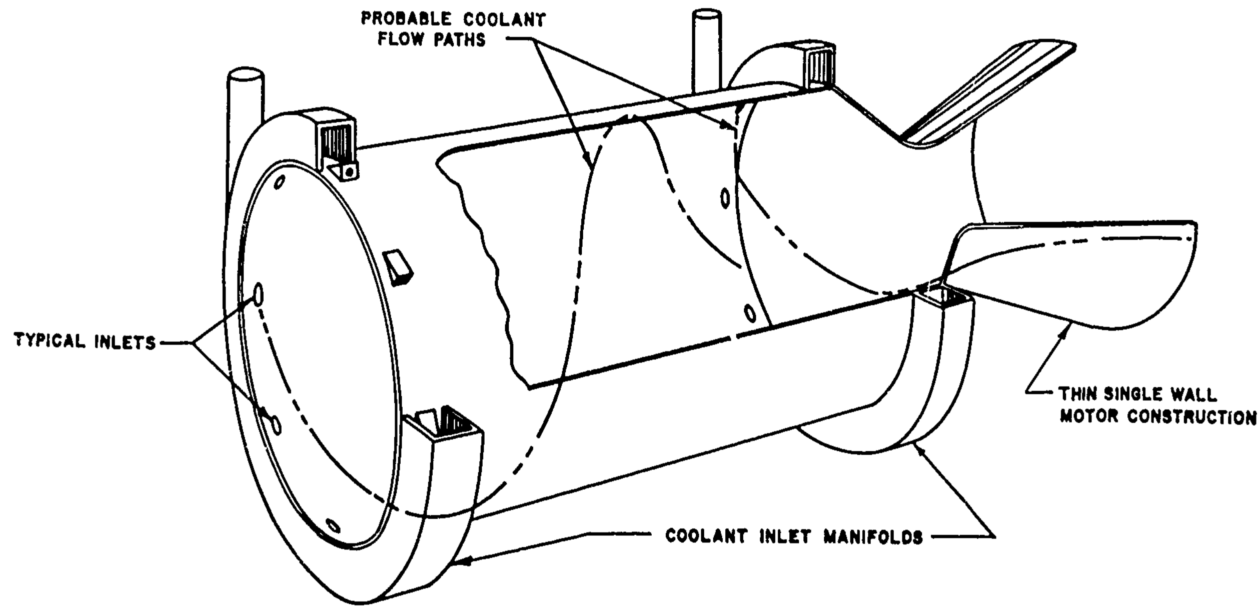

Figure 8. Goddard tangential inlet film cooling.

The primary advantage is that the heat transferred to the combustion chamber wall and thence to the cooling propellant, is returned to the chamber. These motors may take several forms. In a bipropellant system one propellant may be used to cool both nozzle and chamber, or one may be used to cool the nozzle and the other the chamber. One agency is developing a system where a third propellant called the coolant is used to cool the nozzle and then is subsequently injected into the chamber.

For the commonly used propellants and chamber pressures of today the regenerative motor has been quite satisfactory, but a doubt exists as to its practicality with the higher chamber pressures and high energy propellants proposed for use with their consequent higher combustion temperatures.

c. Film cooling of motors was satisfactorily demonstrated as long ago as 1935 in R. H. Goddard's sounding rocket. The principle consists essentially in injecting extra fuel or a coolant through openings in the chamber wall, so that a film exists between the hot gases and the chamber wall. The heat of vaporization and insulation provided by the coolant keep the wall temperature down within the safe limits. The subsequent burning or vaporization of the coolant decreases the loss over the propellant flow which would be present if additional action did not exist. Dr. Goddard used a tangential injection of the coolant as shown in Fig. 8.

A combined regeneratively cooled and film cooled motor, similar in principle to the German A-4 or V-2 motor, is shown in Fig. 9. The extra cooling over the

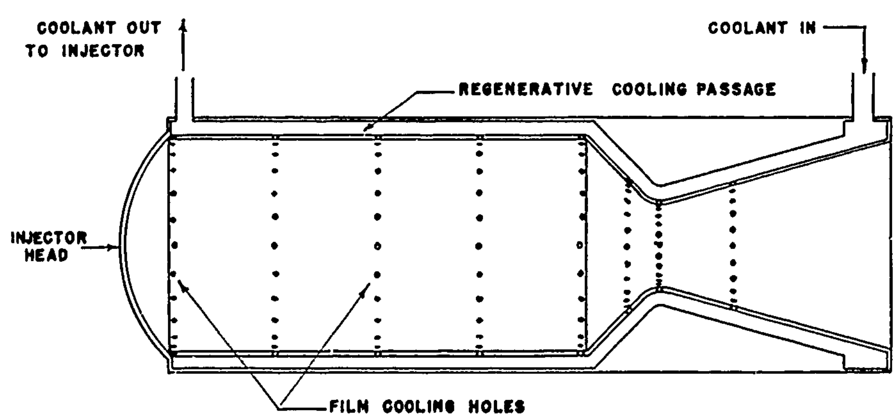

Figure 9. Combined regenerative and multiple hole film cooling.

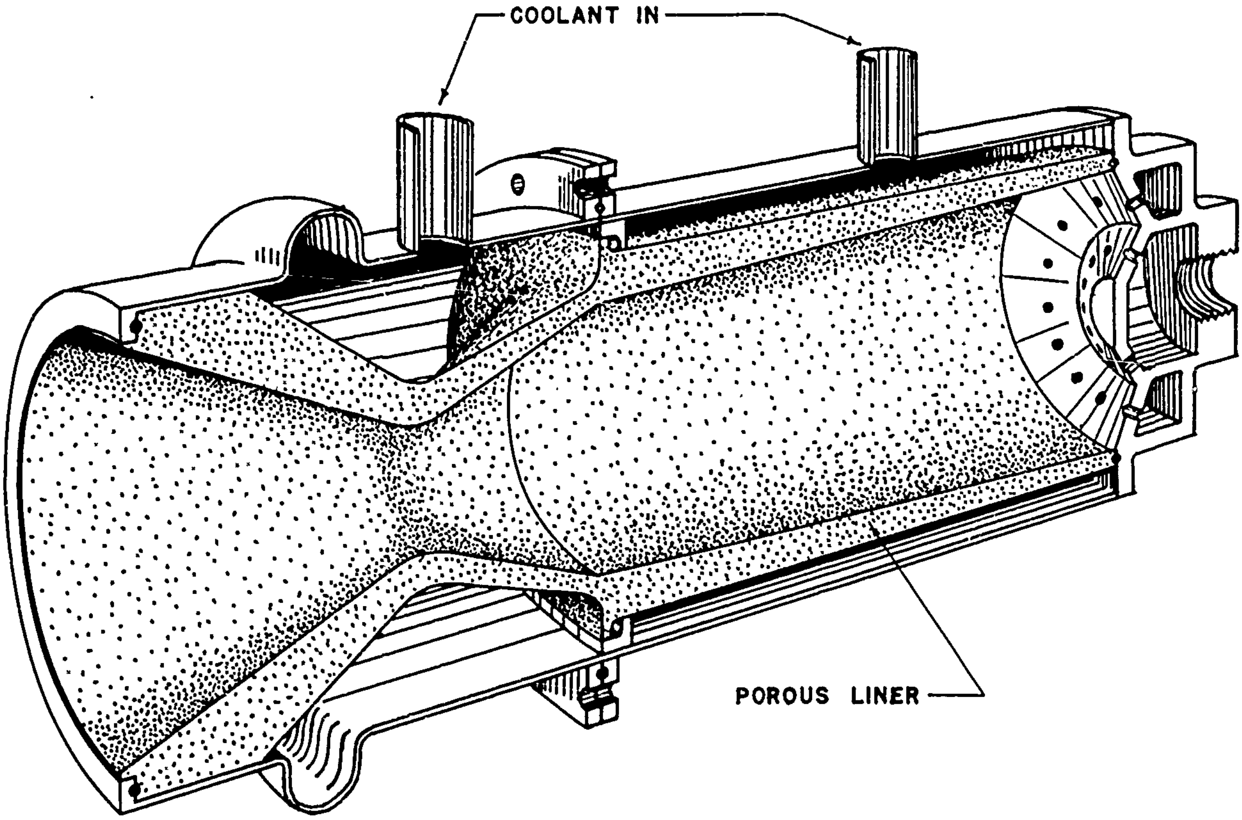

Figure 10. Transpiration or sweat cooled motor.

regenerative capacity is provided by the numerous small holes spaced in rings along the chamber and nozzle.

d. Transpiration or *sweat* cooling to date has not been fully developed. It consists essentially of a porous wall for the inner wall or combustion chamber of the rocket motor (Fig. 10).

The wall may be either a porous metal or a ceramic. Cooling is effected by the passage of one of the propellants or a coolant through this wall, so that its passage cools the wall and its evaporation upon entering the chamber further cools the wall or takes up the heat of combustion before it reaches the wall. Evaporation may take place before the passage is complete, but at present this is considered undesirable. An endothermic decomposition of a propellant as it passes through the wall would also aid in the cooling and would thus considerably reduce the flow required.

There is much to be learned about the construction, strength, permeability, and porosity required for these liners. Obviously the requirements will be different for different propellants. This system of cooling offers many possibilities and must be pursued intensively to establish more complete information.

5. INJECTION. The problems of injecting the propellants into the combustion chamber and properly mixing them are as manifold as there are possible propellant combinations. The overall problem may be divided into two sections: the physical or mechanical characteristics of the injector, and the operating conditions. These sections will be broken down in outline form to indicate the variables.

a. Physical or mechanical characteristics of the injector.

1. openings:
   a. size, shape, and number of individual openings
   b. form of approach passage
   c. porous wall
2. Orientation:
   a. axial location in motor such as head or nozzle end
   b. radial location in motor
   c. length of stream between injector face and point of impingement
   d. angularity with reference to motor axis
   e. angularity between impinging streams

f. angularity between resultant of mixed stream and motor axis
g. distance between impingement point and motor wall or turbulizer

b. Operating conditions.
1. propellant combination:
   a. spontaneous or non-spontaneous
   b. speed of reaction
   c. physical properties
      (1) liquid or gas
      (2) temperature
      (3) density
      (4) viscosity
      (5) heat capacity including latent heat of vaporization
2. flow conditions:
   a. injection pressure or feed pressure
   b. injection differential pressure
   c. flow rates
   d. mixture ratio
3. chamber conditions:
   a. chamber pressure
   b. chamber temperature (also radiation)
   c. density and viscosity of the burning gases.
   d. turbulence and velocity (direction) of burning gases
   e. form of chamber
   f. heat transfer

Needless to say, the taking into account of all these variables to form the optimum injector would be a tremendous task. It is important to note that the present proximity of jet velocity to the theoretical value indicates relatively high efficiencies for the common types of injectors now in use. Figures 11 and 12 indicate schematically the two principal types. Figure 11 represents the intersecting stream type and Fig. 12 represents the intersecting spray type. Both types have been used satisfactorily with both spontaneous and non-spontaneous propellants.

There are three additional factors which are important in injector design.

a. Combustion stability must be considered. The phenomenon called "chugging" may be attributed to injector design, although this has not been positively established.

b. The motor heat transfer may be substantially affected by the orientation of the injector holes or the location of the oxidizer or fuel with respect to the chamber wall.

c. Safety in injector design requires the prevention of mixing of the propellants within the injector due to dribbling around the orifices or through leaks at construction joints. This is especially serious with spontaneous propellants, since an explosion is almost certain to result.

No serious effort has been applied to injection from other than the forward end of motors.

6. Ignition. The use of spontaneously ignitible propellants has eliminated the need for a combustion initiator and hence simplified motor design to this extent. But numerous useful combinations are not spontaneously combustible, such as liquid oxygen-alcohol, and consideration must be given to the problem.

The basic principle is to provide a sufficient volume of high temperature gases or a surface at a temperature

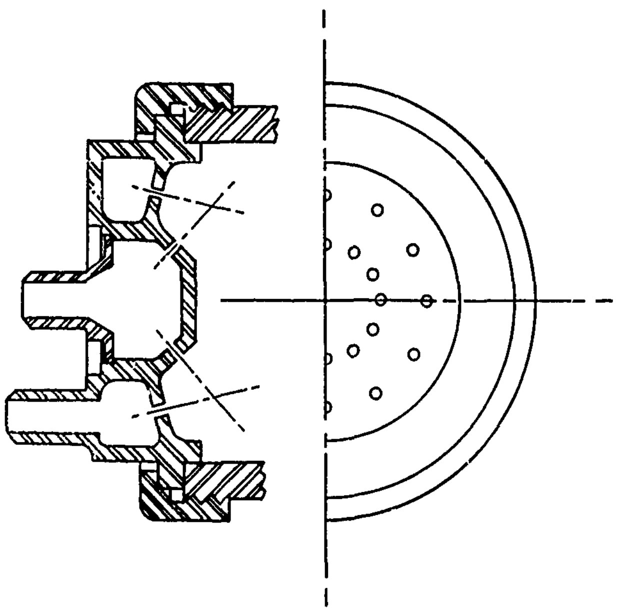

Figure 11. Multiple hole injector.

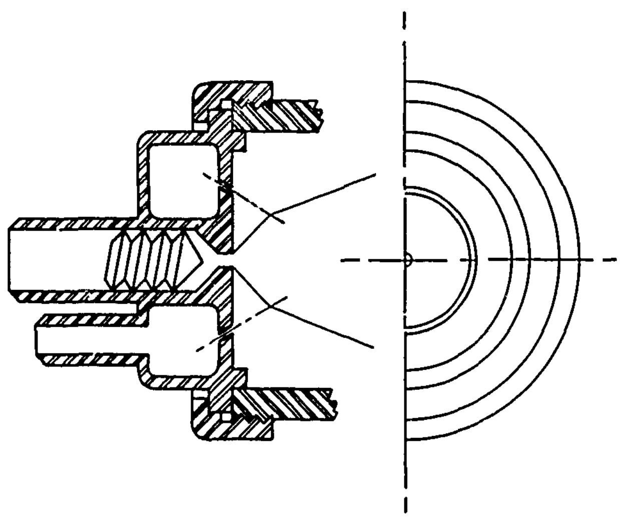

Figure 12. Spray injector.

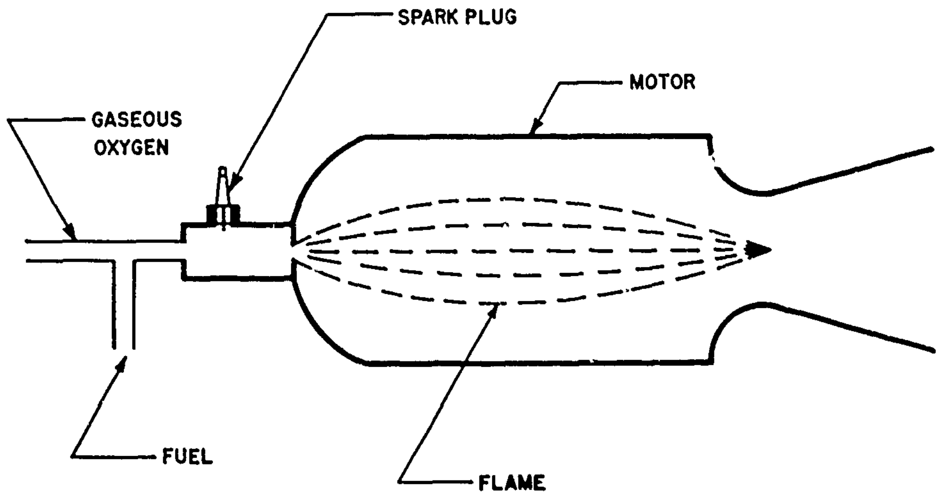

Figure 13. Gaseous oxygen-alcohol igniter.

high enough so that the temperature of the initial flow of propellants will be raised to the ignition point before the heat capacity of the incoming propellant absorbs the available heat. There are several possible methods of accomplishing this:

a. Hot gases or combustion products.
  1. liquid or gaseous propellants
  2. solid propellants

b. Spark (electrical).

c. Hot surface (electrically heated coil).

d. Catalysis.

It is possible to combine two or more of the above methods. One successful type of igniter developed by Reaction Motors, Inc., consists of a small chamber using a spark plug to ignite a small flow of gaseous oxygen and liquid alcohol. This assembly is mounted in the center of the injector head, as shown in Fig. 13, and fills a large portion of the combustion chamber with flame just prior to the entrance of the main flow of liquid oxygen and alcohol.

Spark ignition by itself is usually insufficient to handle the ignition of motors of any substantial size.

Solid propellant igniters such as hot-wire-initiated black powder, which in turn ignites a slow-burning propellant, have been successfully used with nitromethane and liquid oxygen-alcohol or gasoline motors. The main objection is their one-shot nature.

The use of hot surfaces for ignition has not been seriously developed. This is due probably to the problem of prevention of melting of such a surface if it remained in the chamber after combustion started.

Catalytic ignition has been developed mainly by German agencies for use with hydrogen peroxide. Sodium and calcium permanganate were the primary developments. The problem there is simply to provide sufficient catalyst and sufficient exposed area to cause complete decomposition of the peroxide. In cases where a fuel is used with the peroxide, the fuel entrance was delayed until the initial decomposition of the peroxide had been started by the catalyser.

In practically all cases it is necessary to provide a slow build-up of flow of the main propellants to allow the igniter to function properly and prevent an accumulation of propellants before combustion starts. This is also true with spontaneously ignitible propellants. Since there is no back pressure in the chamber, and if full feed pressure is applied, the flow rate will be above normal and a hard start will result. In many cases this will be sufficient to damage or destroy the motor.

7. Life and Operation. Once a rocket motor reaches a state of thermal equilibrium its duration is limited only by the amount of propellant available. Present-day regeneratively cooled motors have demonstrated, without failure, an accumulated operation time of over twenty hours.

The starting and stopping of motors designed for repeated use imposes certain problems. For example, in a regenerative motor the inner chamber may collapse on shut-down when the full supply pressure on the propellant in the cooling jacket is coupled with the water hammer effects of a too rapid shut-down. Also on shut-down, when the propellants are run out and one propellant is used for a coolant, the cooling ceases for a short period while the motor is still running. This is likely to cause overheating, melting, erosion, or complete collapse of the combustion chamber liner and nozzle. In the case of nitromethane regenerative motors, this short period heating may cause decomposition of the nitromethane and result in a violent explosion.

In the case where the system is expendable and no human operator is required, weight may be saved by cutting safety factors to a minimum and eliminating the strength and cooling capacities required at shutdown. If ignition is required, one-shot type igniters may be used, which are blown clear as the motor starts, thus simplifying the fabrication problem and further reducing the weight.

8. Operating Altitude. As discussed previously and shown in Figure 3 for each given chamber pressure and external pressure, there exists an optimum nozzle, whose ratio of exit to throat area gives maximum thrust. Thus for a constant chamber pressure a rigid nozzle can be designed correctly for only one altitude. Where the altitude varies continuously the nozzle must be designed for some average altitude based on time conditions. For example, using the theoretical curves for C_F, if a nozzle were designed for maximum efficiency at 25,000 ft., with a P_c of 300 p.s.i.a., the decrease from maximum possible thrust coefficient at sea level would be 5%, and approximately 3% for 50,000 ft. Preliminary tests at Aerojet Engineering Corporation indicate that only about two-thirds of this loss is actually encountered.

9. Fabrication. The general adoption of the cylindrical form indicates the influence of fabrication on motor design. Permanent assembly by welding, which virtually makes repair impossible, has generally been adopted because it provides a simple lightweight leak-proof joint. The importance of leak-proof joints cannot be overestimated in rocket motors. Inadvertent mixing of spontaneous propellants in confined areas results in explosions, or a small leak of hot combustion chamber gases often quickly erodes a hole large enough to destroy the motor.

Other methods of construction of motors have not been seriously investigated. One agency, however, has attempted electro-forming (or electro-plating) metals on a low melting point pattern electrode. The electrode is melted out when the plating has reached sufficient thickness. This system thus lends itself to odd shapes and geometrical arrangements. No complete motors have been made to date by this process, but its development is still underway. Metal spraying might also be adaptable for this purpose.

D. Propellant Feed Systems

The problem of supplying the propellants to the rocket motor may be divided into two general classes: pressurized systems, and pumping systems.

1. Pressurized Systems. The essential components of the pressurized rocket system are the gas storage tank, pressure regulator, propellant tanks, control valves, and rocket motor. Figure 14 shows a schematic

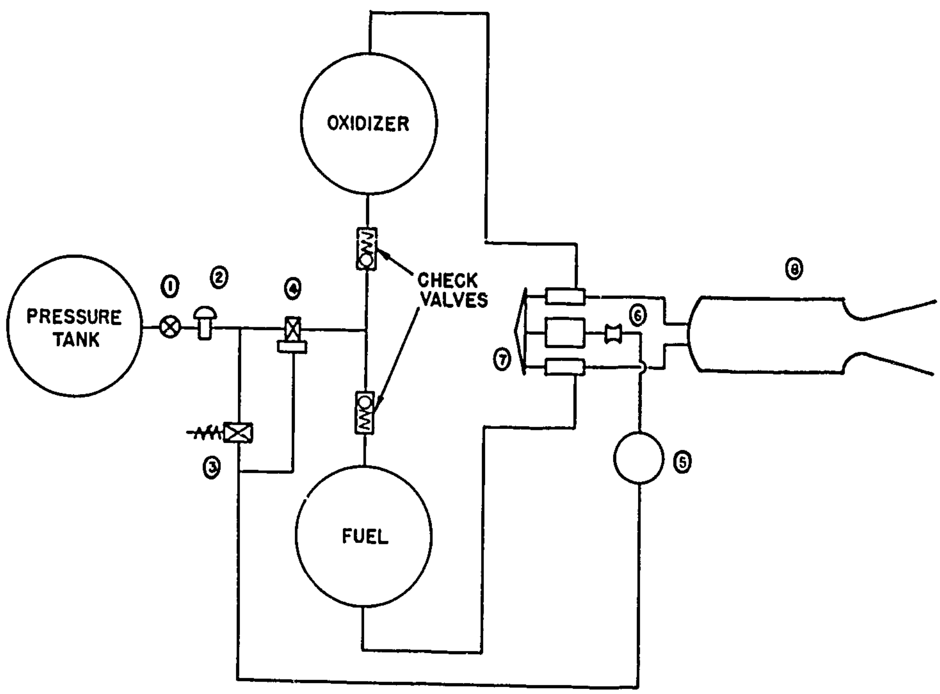

Figure 14. Pressurized rocket system.

diagram of a typical system, including minor but necessary items such as check valves, etc. The range of usefulness of the pressurized rocket system is from five to fifty seconds. In the upper time-region this system is heavier and bulkier than the equivalent pumping system, but its inherent simplicity and lower cost warrants its use.

The gas pressure tank is filled with an inert gas from 1,800 to 2,000 lbs./sq. in. and the volume is such that it will completely expel all of the propellants from their respective tanks at their designed feed pressure. Usually this is of the order of 400 lbs./sq. in. for pressurized systems. The operation may be described as follows, in Figure 14. Hand valve (1) is opened, allowing the high pressure to pass to the pressure regulator valve (2). The motor operation is controlled by electric valve (3). Opening this valve allows pressure to valve (4), which in turn allows the reduced pressure to enter the propellant tank, and simultaneously the pressure enters hydraulic accumulator (5), which forces oil through restricter (6), and then into the actuating cylinder of the main control valve (7). The propellants are then free to enter the combustion chamber (8). The oil accumulator and restricter valve in the main control valve actuator line are for the purpose of controlling or restricting the propellant flow during the starting period. As was described in the section on ignition, this is necessary to prevent hard starts. Stopping of the system is readily effected by closing valve (3), which vents the lines between valves (4) and (7), and hence allows them to close.

The selection of the propellants, size of motor, chamber pressure, and duration, determines the size of the required pressurizing gas tank.

The selection of a pressurizing gas depends on several factors, the most important of which is its inertness with respect to the propellants, and second, availability and handling characteristics.

Air is, of course, the most readily available gas, but it is unsafe for use with nitromethane or gasoline, and in some situations where high temperatures are encountered, with aniline. Air with hydrocarbon vapors present might also be unsafe over acid at high pressure.

Nitrogen is most commonly used, since it may safely be compressed over all of the common propellants. It is, of course, not as readily available as air and has the additional disadvantage of tending to condense and go into solution with liquid oxygen. For this reason helium is used in many experimental setups with liquid oxygen. The lack of availability and cost, however, prevent widespread use of helium.

Other inert gases have been tried, such as CO_2. It is generally not satisfactory, owing to its tendency to freeze in the pressure regulator and to dissolve in the propellants, causing a loss in pressure.

An inert gas generator for field service has been built by the Aerojet Engineering Corporation for the U. S. Navy. The inert gas is produced by removing the oxygen from the air by burning it in a gasoline engine. The constituents of the gas after purification and water removal consists of approximately:

Nitrogen (N_2)	86%
Carbon dioxide (CO_2)	12.5%
Carbon monoxide (CO)	1.29%
Oxygen (O_2)	2.00%
Water (H_2O)	.01%

It may be possible to improve the efficiency of pressurized systems by heating the gas after it passes through the regulator. Cooling a small portion of the motor with the gas could be used to provide the necessary heat.

2. Pumping Systems. The essential components of a pumping system are propellant tanks, pumps, source of power for the pumps, control valves, and the rocket motor or motors. The pumping system is dependent on the size or thrust of the motors but is independent of the duration. In general, when a rocket system is required to operate for longer than fifty seconds, it is desirable to use pumps.

The power required to operate pumps can best be understood by an example. Approximately 15 h.p. is required to supply the propellants to a 1000-lb. thrust rocket motor with a feed pressure of 500 p.s.i.

The selection of the type of pump to be used depends on several factors:

a. type of drive available,
b. propellant flow required,
c. adaptability of propellant to pump type,
d. suction pressure requirements,
e. delivery pressure required.

Since proper evaluation of the variables depends primarily on the types of drives and types of pumps available, a brief discussion of the known types is presented below.

a. Types of Pumps:

1. *Gear pumps* are of the positive displacement type. They depend on close clearances to avoid slippage, and hence are more satisfactory for liquids of high viscosity. Operation with nitric acid is undesirable because stainless steel, which generally must be used with acid, tends to gall on close clearances. With the low viscosity liquids commonly encountered in rocket systems an efficiency of 40% is difficult to ex-

ceed with a delivery pressure of 500 p.s.i. In general, when the required volume of flow is above 25 g.p.m. and a high viscosity (above 10 centistokes) and low corrosivity are present, the gear pump will be satisfactory. Rotational speeds of approximately 2,000 r.p.m. are most common.

2. *Piston pumps* are suitable for low viscosity fluids requiring high delivery pressure. The fine clearances required again make this type of pump undesirable for use with acid. Efficiencies as high as 90% may be obtained. However, piston pumps for high flow rates are considerably heavier than other type pumps to be discussed. Rotational speeds are generally between 2,000 and 4,000 r.p.m.

3. *Turbine and propeller pumps* generally are suited best for low pressure and low flow rates. However, they may be suitable for booster pumps on centrifugal pumps to prevent cavitation.

4. *Centrifugal pumps* are the most suitable for all around use with liquid propellants for rockets. The larger allowable clearances make construction practical with stainless steel and hence satisfactory for acid pumping. The efficiency is not appreciably decreased with wear, and they are lighter than other types for supplying the necessary flow rates and delivery pressures required in rockets. Efficiencies of the order of 65% to 85% may be realized with proper suction pressures. The high operating speeds 10,000 to 30,000 r.p.m. make it suitable for use with a high speed turbine. Liquid oxygen has been successfully pumped at 12,000 r.p.m. and acid and aniline up to 30,000 r.p.m. Figure 15 shows a pair of centrifugal pumps mounted with their turbine drive. This assembly weighs 4.5 lbs.

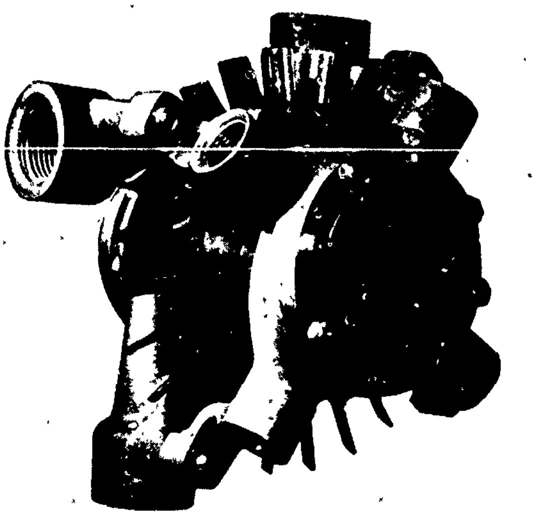

Figure 15. NAMTC rocket exhaust-driven turbine and pumps.

and is designed for a 6,000-lb. thrust white fuming nitric acid-alcohol system. The pumps and turbines were designed by James Coolidge Carter, of Pasadena, California, for the Naval Air Missiles Test Center, Pt. Mugu. It is planned to drive the turbine with the edge of the exhaust of the four 1,500-lb. thrust motors.

b. Types of Pump Drives:

1. *Accessory drive* from the main engine of the aircraft. Such a drive may be considered in the case of jet assisted takeoff or superperformance of aircraft. This drive may be from either a reciprocating engine or a turbojet. It has a basic fault in that failure of the main engine precludes the use of the assisting rocket and hence all safety value is lost. The use of this type of drive has been considered in this country but not developed primarily for the reason mentioned above. The BMW Company of Germany, however, developed such an arrangement for use on the ME-262 airplane with BMW turbojets.

2. *Gas bleed drive from turbojet* to drive a turbine. In this case either air may be taken directly from the turbojet compressor or hot gases from one of the combustion chambers of the turbojet. The difficulty or fault with this idea is the same as that for the auxiliary drive systems. No serious attempt has been made to develop this system, although it may be developed as a means of starting the turbine for a turbopump rocket system. This is under consideration by the Curtiss Propeller Division of the Curtiss-Wright Corporation.

3. *A wind propeller or turbine* has been proposed for a pump drive. This system would only be useful in the subsonic range and the complications of variable pitch blades to maintain constant output for even a moderate range of speeds would be difficult in proportion to the end result. Even then the fuel consumption in overcoming the additional drag would be approximately twice that for a turbopump installation.

4. *The liquid propellant combustion pot and gas turbine*, commonly known as a turbopump system or turborocket, has the greatest number of advantages for many applications. The power to drive the turbine is derived from a small combustion chamber which uses either the same propellants as the main rocket system or it may use a separate fuel or oxidizer or both. In some cases a diluent, such as water, is required to reduce the gas temperature before passing through the turbine blades. The propellant flow required for this drive may vary from 1.5% to 3% of the total flow rate of the system, depending on the efficiency of the design. The advantage of this system of turbine drive for pumps is in the high speed which

lends itself readily to direct drive of centrifugal pumps. Efficient design can produce a power plant which will weigh under one lb./h.p.

The A-4 German missile represents a turbopump system where a separate set of propellants is used in the combustion pot from that used in the main rocket. Hydrogen peroxide and a catalyst are used in the turbine combustion pot while liquid oxygen and alcohol are used in the main system.

Figure 16 below represents schematically a simplified system using common propellants. Spontaneously combustible propellants are assumed and all timing and pressure switches are left out to avoid complication in explanation. A small tank (1) is required for low pressurization to prevent pump cavitation. This will vary from 10 to 30 p.s.i., depending on the propellants used. Pressure regulator (2) reduces the pressure to the required value and electrically operated valve (3) is used to release this pressure to the propellant tanks (4) and (5). Valves (6) and (7) may then be opened to allow the propellants to enter the combustion pot (8). This is sufficient to start the turbine (9) and pumps (10) and (11). When the discharge pressure reaches a certain value, a pressure switch opens valves (12) and (13) and closes valves (6) and (7). The turbine and pumps will then come up to speed and the necessary propellant is by-passed to prevent overheating. With prestart readings up to pressure the operator may open valves (14) and (15) and allow the propellants to enter the rocket motor (16). Stopping of the system requires closing of valves (3), (14), and (15), and the opening of vent valves (17) and (18) on the propellant tanks.

5. *Exhaust turbopump drive* is also a turbine for driving the pumps but in this case the turbine is driven by the exhaust gases of the main rocket motor or motors. Figure 17 shows a test setup at the Naval Air Missiles Test Center for an exhaust turbopump drive with pumps mounted for operation with mixed acid and monoethylaniline on a 220-lb. thrust motor. This type of drive eliminates the necessity for a special combustion pot and its controls but adds the requirement of a starting system to get the turbine and pumps up to a speed sufficient to start the motor. This starting system, however, need not be part of the vehicle. Figure 18 represents a simplified schematic drawing of this type of system and illustrates the decrease in controls required for operation in comparison to the system described in Fig. 16. Pump suction head pressurization and starting air tank (1) supplies gas to pressure regulator (2), and valve (3) releases the reduced pressure to propellant tanks (4) and (5) and to starting air valve (8). Propellant valves (6) and (7) may then be opened simultaneously with starting

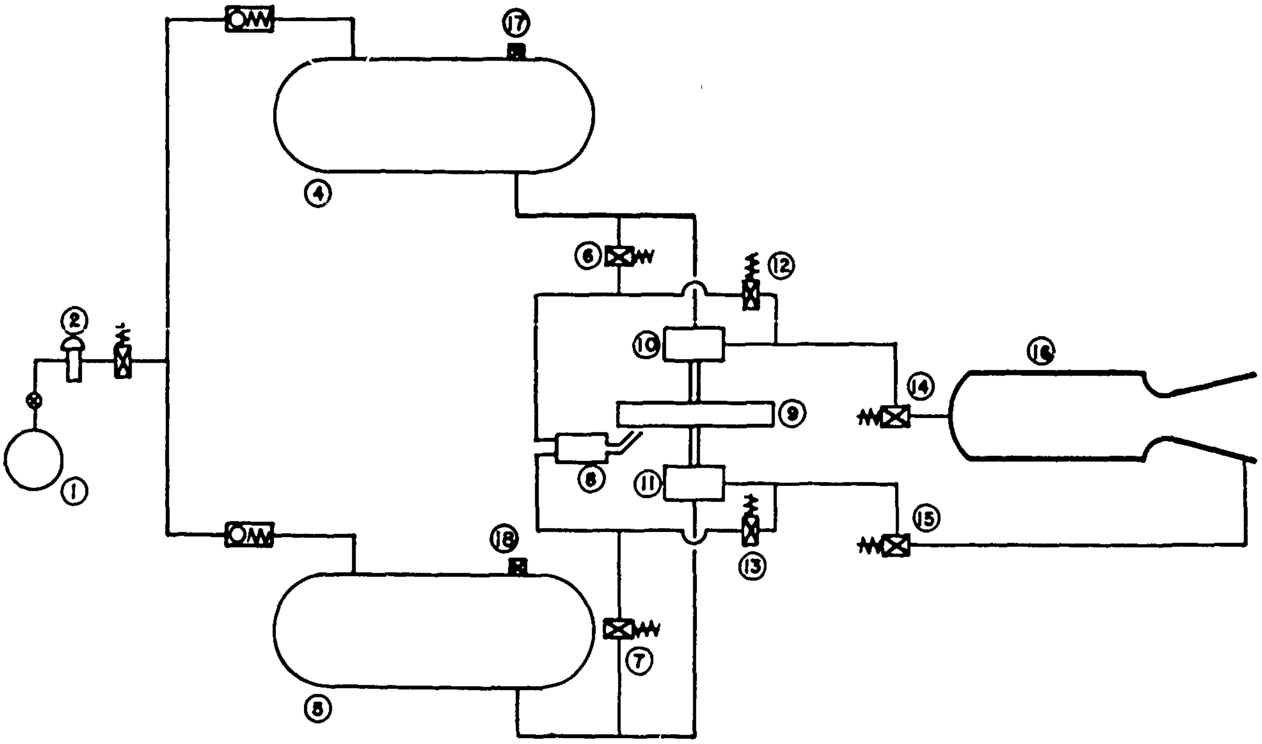

Figure 16. Turbopump rocket system.

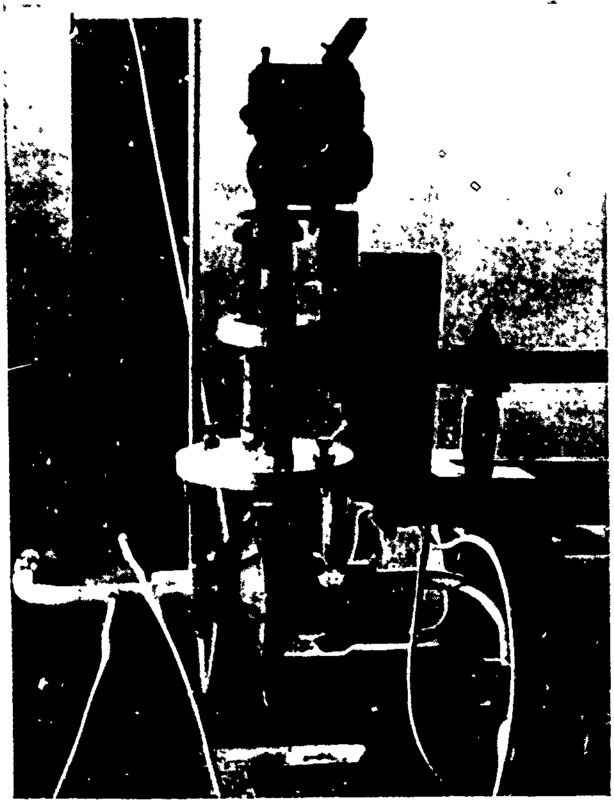

Figure 17. CML-4N exhaust turbopump test setup.

air valve (8). As the turbine comes up to speed, the pumps (10) and (11) start to supply propellants to motor (12). The motor exhaust then starts to drive the turbine and a chamber pressure operated switch may be used to close valve (8) to stop the starting air. Stopping is accomplished by closing valves (3), (6), and (7) and opening valves (13) and (14) to vent the tanks.

The exhaust turbopump drive was first proposed and tested by Dr. Goddard, and development has since been carried out by the Naval Air Missile Test Center and the Eclipse Pioneer Division of the Bendix Aviation Corporation in conjunction with Reaction Motors, Inc. It is felt that this system may be lighter than the equivalent turbopump rocket, provided adequate cooling of the turbine can be maintained at altitude.

6. *An auxiliary engine drive* for rocket propellant pumps has been developed for a 6,000-lb. thrust acid-aniline system by the Aerojet Engineering Corporation under Navy sponsorship. Small aircraft engines are available in the horsepower range required for such systems, but these engines are not supercharged and hence are suitable for low altitude work only. These systems are inherently heavier than the turbopump type systems, because the driving engines usually weight 2 to 3 lbs./h.p.

7. *The aerotojet and the centrojet* are schemes for mounting the rocket motors on a shaft in such a manner as to cause rotation of the shaft when the motor is operating (Fig. 19). The rotating shaft is then used to drive pumps. A 2,000-lb. rocket system has been built, utilizing this principle, by the Aerojet Engineering Corporation under Army Air Forces sponsorship. Serious difficulties, however, have been encountered in connection with combustion and injection in

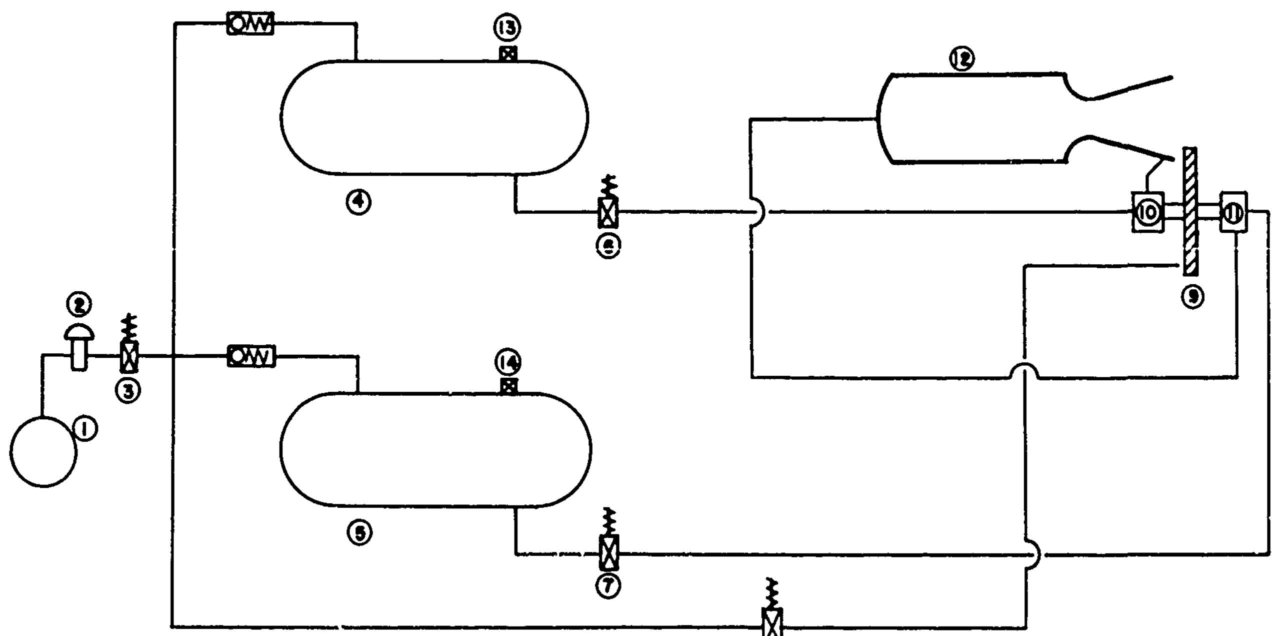

Figure 18. Exhaust turbopump rocket system.

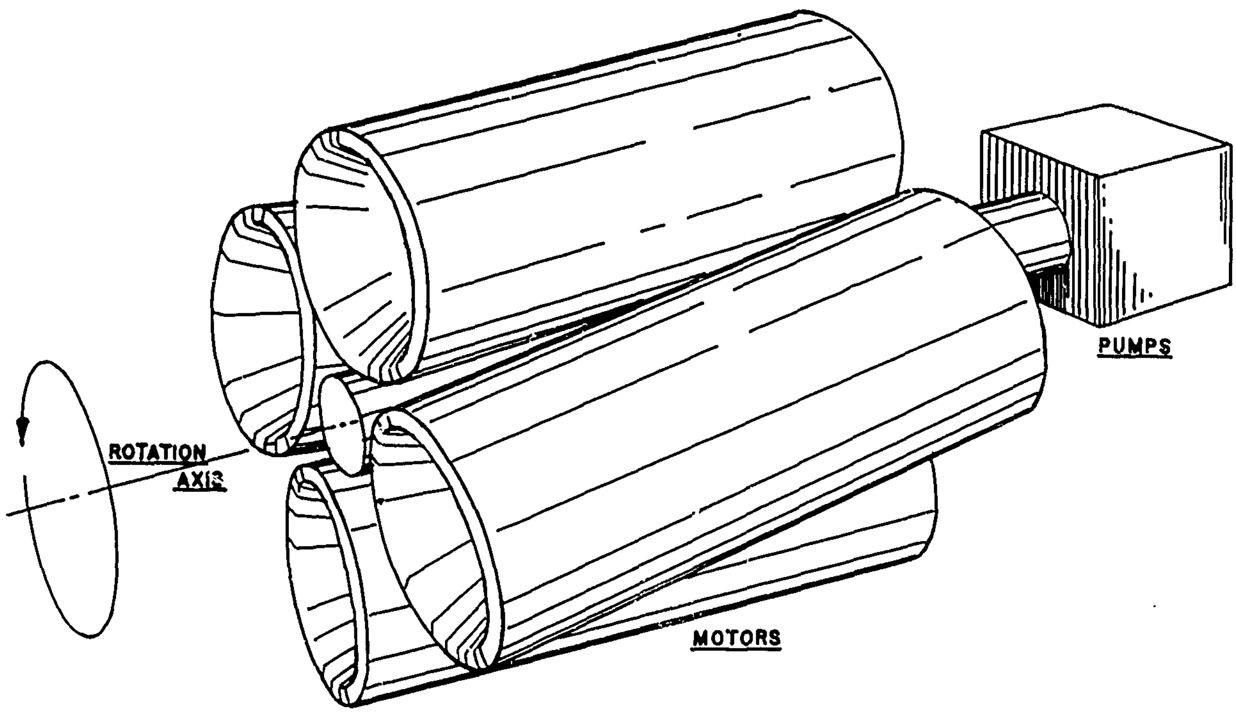

Figure 19. Principle of Aerotojet.

the rotating chambers. The problem has not been satisfactorily solved at this time and it is felt that this system will be inherently heavier than other means of pump drive.

The centrojet is essentially an extension of the aerotojet principle, in which a single chamber rotates at higher speeds and hence the centrifugal action of the propellant in the lines extending from the pump outward to the motor is utilized in raising the delivery pressure. This is accomplished by two or more skewed nozzles on the chamber so that rotation results. The inherent difficulties of the aerotojet are still present, however.

E. Liquid Propellants for Rockets

The choice of liquid propellants for a given rocket system depends upon several factors: (1) availability, (2) specific impulse, (3) density, (4) properties (chemical and physical), and (5) safety and handling characteristics. The availability, safety, and handling characteristics are factors which will influence the choice in varying degrees, depending on the urgency of the situation. Since urgency will vary from time to time, it must be kept in mind constantly.

The performance parameters of liquid propellants for rockets are essentially the same as those described under Section C, above, on Principles and Design of Liquid Propellant Rocket Motors. The principal factors in increasing the characteristic jet velocity were shown to be high combustion temperature and low average molecular weight of exhaust gases. The characteristic velocity c^* may be used to compare various propellant combinations, but it has become the general practice to use the specific impulse I_{sp}. In addition, the density impulse I_d has been found to be useful, since it gives an indication of the volume required to handle the propellants. This is particularly important in missile applications where drag must be considered, but it must be remembered that it has no fundamental significance with regard to motor performance. Density impulse is determined simply by multiplying the density of the propellant combination, in the c.g.s. system, by the specific impulse, thus:

$$I_d = I_{sp} \times \rho$$

A summary is presented below in Table I of the most common or most promising propellant combinations thus far investigated on either a theoretical or an experimental basis. The systems are listed by oxidizer for ease of reference. This table, with certain modifications, was made up from references (23) and (31). Table II gives some of the physical and chemical properties of the more well-known propellants (19).

A more detailed account of the field of rocket propellants will be found in Volume I, Part 2, *Fuels*, of this survey series.

Table I. Liquid Rocket Propellant Performance Data.

	Mixture Ratio Oxidizer/Fuel	Bulk Density	Chamber Temperature °R	Abs. Chamber Pressure in Atmospheres	I_{sp}	I_d
Oxidizer-Flourine						
Hydrazine	1.186	1.061	6,970	20.0	292	310
Hydrazine	2.371	1.078	9,500	20.0	312.5	337
Hydrogen	18.85	.637	10,210	20.0	352	224
Hydrogen	9.42	.366	8,530	20.0	371	136
Hydrogen	6.28	.270	6,296	20.0	352.5	95
Hydrogen	4.72	.221	5,586	20.0	357	79
Lithium	2.738	.861	15,020	20.0	351	302
Lithium	1.368	.762	5,679	20.0	259	197
77.6% Lithium; 22.4% Hydrogen	2.122	.475	7,134	20.0	339	161
63.3% Lithium; 36.7% Hydrogen	1.731	.342	5,420	20.0	335	115
53.6% Lithium; 46.4% Hydrogen	1.464	.274	4,700	20.0	346	95
Oxidizer-Hydrogen Peroxide						
54.8% Ethylene Diamine; 45.2% H_2O	1.243	1.174	3,140	20.4	196.3	231
Hydrazine	1.415	1.237	4,890	20.4	249	308
Methyl Alcohol	3.68	1.190	4,590	20.4	225	268
Nitromethane	1.50	1.27	5,350	20.4	227	289
93.4% Nitromethane; 6.6% methyl alcohol	0.333	1.11	4,950	20.4	226	250
Nitromethane; 35% Nitroethane	—	1.155	—	20.4	218	253
Oxidizer-Mixed Nitric Acid						
Monoethylaniline	4.60	1.396	5,060	23	210	293
Oxidizer-Red Fuming Nitric Acid						
Aniline	3.00	1.390	5,525	20.4	220.5	307
Ethyl Alcohol	2.90	1.26	5,330	20.4	218.0	274
Hydrazine	.772	1.180	4,510	20.0	243	288
Oxidizer-White Fuming Nitric Acid						
Furfuryl Alcohol	1.90	1.355	4,753	20.0	213	287
Tonka	—	1.354	—	—	226	306
Aniline	3.00	1.349	5,360	20.4	219	296
Oxidizer-Nitrogen Dioxide						
Acetylene	2.94	1.09	6,220	23.0	256	279
31.7% Acetylene; 68.3% Ammonia	2.24	1.12	5,530	23.0	244	274
Propane	4.153	1.125	5,580	23.0	238	268
Oxidizer-Liquid Oxygen						
Acetylene	1.230	.519	6,474	20.0	266	138
48.6% Aluminum; 51.4% Octane	2.260	1.12	6,060	20.0	247	277
Aluminum Borohydride	2.668	.860	5,890	20.0	250	215
Aluminum Borohydride	1.334	.737	7,510	20.0	288	212
Aluminum Trimethyl	1.000	.940	6,160	20.0	250	235

Table I. Liquid Rocket Propellant Performance Data. (Continued)

	Mixture Ratio Oxidizer/Fuel	Density Bulk	Chamber Temperature °R	Abs. Chamber Pressure in Atmospheres	I_{sp}	I_d
Oxidizer-Liquid Oxygen						
Ammonia	1.41	.978	5,410	20.4	255	250
30.0% Beryllium; 70.0% Octane	1.75	1.02	7,120	20.0	253	258
Beryllium Dimethyl	1.23	—	6,160	20.0	270	—
Boron	2.33	1.497	22,012	20.0	278	416
64.2% Boron; 35.8% Hydrogen	2.12	.365	5,892	20.0	330	120
Diacetylene	1.28	—	7,640	20.0	271	—
Diborane	3.470	.852	7,520	20.0	240.5	206
Diborane	1.734	.728	7,560	20.0	294	214
Dipropargyl	3.076	.961	5,654	20.0	247	237
Ethane	1.070	.760	1,910*	20.4	180	137
Ethyl Alcohol (100%)	1.50	.966	5,720	20.4	243	235
75% Ethyl Alcohol; 25% Water	1.275	.992	5,530	20.0	233	231
Gasoline	2.51	.978	5,930	20.4	242	236
48.8% Gasoline and 51.2% Nitrogen	1.198	.931	5,290	20.4	221.5	207
Hydrazine	1.00	1.072	5,900	20.4	257	276
Hydrogen	4.00	.284	5,500	20.0	353	100
Lithium	1.152	.662	13,500	20.0	318	210
77.5% Lithium; 22.5% Hydrogen	.889	.286	4,470	20.0	296	85
Lithium Borohydride	2.938	.90	6,970	20.0	259	233
Lithium Borohydride	1.469	.83	8,810	20.0	306	254
Methane	3.00	.804	5,500	20.0	254	204
Methyl Alcohol	1.25	.955	5,640	20.0	237	226
Methylamine	2.067	.985	6,100	20.4	266	262
Nitromethane	.0766	1.139	5,160	20.4	225.5	256
Ethylene	1.15	.774	—	20.4	236	182
Oxidizer-Water						
Diborane	2.00	.706	—	20.4	200	141
Monopropellants						
Diethylene-glycol Dinitrate (100%)	—	1.483	4,590	20.4	215	319
Nitro Benzine (10%)	—	1.181	3,960	20.4	204	241
Ethyl Nitrate (100%)	—	—	3,530	20.4	203	—
Hydrogen Peroxide (100%)	—	1.463	2,258	20.4	146	213
87% Hydrogen Peroxide; 13% Water	—	1.381	—	20.4	126.3	175
80% Methyl Nitrate; 20% Methyl Alcohol	—	—	4,370	20.4	221	—
70% Nitroglycerine; 30% Nitrobenzine	—	—	4,950	20.4	217	—
Nitromethane	—	1.139	4,590	20.3	222	247
90% Nitromethane; 10% Nitrobenzine	—	1.18	3,980	20.4	211	248
83% Nitromethane; 17% Nitrobenzine	—	1.123	3,940	20.4	206	232

*This value appears too low.

A study of Table I indicates that the following propellant combinations are superior on the basis of density impulse and reasonable chamber temperature. It is also interesting to note that none of the so-called high energy fuels are present on the list. This table does not represent a complete evaluation of all the propellants but rather an indication of a halting place to take stock of the overall situation.

Propellant Combination	I_d
Hydrazine-Flourine	337
Hydrazine-Hydrogen Peroxide	307
Hydrazine-Red Fuming Nitric Acid	288
Hydrazine-Liquid Oxygen	279
Aniline-Red Fuming Nitric Acid	307
Aniline-White Fuming Nitric Acid	294
Monoethylaniline-Mixed Acid	293
Nitromethane-Hydrogen Peroxide	276
Methyl Alcohol-Hydrogen Peroxide	278
Tonka-White Fuming Nitric-Acid	306

Table II. Characteristics of Liquid Rocket Propellants.

Propellant	Chemical Formula	Melting Point °F	Boiling Point °F	Density lbs./cu. ft.	Toxicity	Corrosiveness to Metals	Availability in U.S.A.	Special Handling Characteristics
Liquid Oxygen	O_2	—361	—297	71.2	None	None	Good	Vented, insulated containers.
Red Fuming Nitric Acid	$HNO_3 + 6.5\% NO_2$	—51	130	95.5	High	High — Requires pure aluminum or stainless steel	Good	Must be kept clear of all organic matter.
White Fuming Nitric Acid	HNO_3	—44	187	93.6	Moderate	High — Requires pure aluminum or stainless steel	Good	Must be kept clear of all organic matter.
Mixed Acid	$HNO_3(88\%) + H_2SO_4(12\%)$	—40		97.4	Moderate	High — Requires pure aluminum or stainless steel	Good	Must be clear of all organic matter. Forms precipitate with mild steel. Requires filtration.
Gasoline	C_8H_{18}(approx)	—40	155	46.8	Low	None	Very good	No special handling — fire precautions.
Ethyl Alcohol	C_2H_5OH	—175	173	49.3	Low	None	Good	No special handling — fire precautions.
Methyl Alcohol	CH_3OH	—144	148	49.5	Moderate	None	Good	No special handling — fire precautions.
Furfuryl Alcohol	$C_4H_3O.CH_2OH$	—24	340	70.5	Low	None	Good	No special handling — fire precautions.
Aniline	$C_6H_5NH_2$	—21	364	63.6	High	None	Good	Open handling should be avoided, readily absorbed by skin, poisonous.
Monoethylaniline	$C_6H_5NHC_2H_5$	—83	400	60.1	High	None	Good	Open handling should be avoided, readily absorbed by skin, poisonous.
Hydrogen Peroxide (100%)	H_2O_2	—29	306	91.1	Low	None Decomposed by iron	Fair	Vented containers required. Must be high purity aluminum or stainless steel. Attacks organic matter.
Hydrogen Peroxide (87%)	$H_2O_2 + 13\% H_2O$	9.5	280	86.8	Low	None Decomposed by iron	Fair	Vented containers required. Must be high purity aluminum or stainless steel. Attacks organic matter.
Nitromethane	CH_3NO_2	—19.5	214	70.5	Low	None	Potentially good	Somewhat sensitive to shock and elevated temperatures. Avoid copper and mild steel containers.
Ammonia	NH_3	—108	28	38.7	Low	Corrodes copper or brass	Very good	Must be stored under pressure.
Hydrazine	N_2H_4	34.5	237	63.1	Moderate	Attacks lead, aluminum, copper and iron	Very poor	Must be shipped airtight.
Hydrazine Hydrate	$N_2H_4.H_2O$	—40	244	64.3	Moderate	Satisfactory in stainless steel	Very poor	Must be shipped airtight.
Diethyleneglycol Dinitrate	$(CH_2CH_2ONO_2)_2O$	11.3	320	86.8	Vapors produce violent headaches	None	Very poor	Sensitive to heat and shock.

IV. COMPLETED DEVELOPMENTS

Complete detailed information is not readily available for all of the liquid rocket systems developed in this country, but in many cases it can be obtained from the agency concerned. A minimum of information has been selected and tabulated for those units developed under military sponsorship. There is one exception to this in the 800-lb. thrust turborocket of R. H. Goddard, which was developed with the aid of the Daniel and Florence Guggenheim Foundation in 1934 and 1935. Figures 20 through 27 are included to show typical examples of completed developments.

A table covering the major German developments has been included for reference purposes.

The following abbreviations are used in Tables III and IV. Table V covers the current liquid rocket systems under development.

Abbreviations

Agencies:

AAF	Army Air Forces
BMW	Bayerische Motor Works, Germany
Bendix	Eclipse Pioneer Division, Bendix Aviation Corporation
BuAer	U. S. Navy, Bureau of Aeronautics
CVAC	Consolidated Vultee Aircraft Corp.
GALCIT	Guggenheim Aeronautical Laboratory, California Institute of Technology
HWK	Walter Works, Kiel, Germany
NAA	North American Aviation, Inc.
NAMTC	Naval Air Missiles Test Center, Pt. Mugu, California
NEES	Naval Engineering Experiment Station, Annapolis, Maryland
RMI	Reaction Motors, Inc.

Propellants:

ALC	Alcohol
AN	Aniline
B Stoff	Hydrazine Hydrate
$Ca(MnO_4)_2$	Calcium Permanganate
C Stoff	30% Hydrogen Peroxide 57% Methyl Alcohol 13% Water Potassium Cuprocyanide in Solution
GAS	Gasoline
H_2O_2	Concentrated Hydrogen Peroxide
JP-1	Kerosene type turbojet fuel
LIQ.H_2	Liquid Hydrogen
LIQ.O_2	Liquid Oxygen
MA	Mixed Acid (approximately 90% HNO_3, 10% H_2SO_4)
MEA	Monoethylaniline
N_2H_4	Anhydrous Hydrazine
NA	White Nitric Acid
$NaMnO_4$	Sodium Permanganate
NM	Nitromethane
O_2	Gaseous Oxygen
Oxid.	Oxidizer
RFNA	Red Fuming Nitric Acid
Tonka	50% Triethylamine 50% Xylidines
Visol	40% Isopropyl Alcohol 40% Vinyl Ether 2% Water 18% Dopes for controlling ignition rate

Miscellaneous:

A/N	Army-Navy
Airp.	Airplane
Comp.	Compounds
Cont.	Continuous
Corp.	Corporal
Exp.	Experimental
HASR	High Altitude Sounding Rocket
JATO	Jet Assisted Takeoff
Min.	Minutes
Miss.	Missile
P/A	Pilotless Aircraft
Press.	Pressurized
Prod.	Production
Prop.	Propulsion
Var.	Variable Thrust
Wt.	Weight

Table III. Summary of Completed American Rocket Developments.

No.	Thrust	Manufacturer	Designation	Duration in secs.	Propellants	Specific Impulse	Fuel Feed System	Chamber Pressure	Status	Use	Sponsoring Agency	Figure No.
1	50	RMI	A50C1	—	LIQ.O_2-Alc.+H_2O	—	Press.	—	Prod.	Exp. Test	Army	
2	130	RMI-(NEES)	CML-1N	90	LIQ.O_2+Gas.	156 secs	Press.	—	Prototype	Pelican P/A	Navy	
3	200	Aerojet	XCALR-200	300	RFNA-AN	150	Press.	—	4 Built	Aircraft Prop.	Army	
4	200	GALCIT	200	30	H_2O_2-NM-MEA	185	Press.	300	Prototype	Experimental	Army	
5	250	GALCIT	JC-80	30 min.	RFNA-AN	185	Press.	400	Prototype	Experimental	Army	
6	300	Aerojet	—	60 min.	RFNA-AN	—	—	400	Prototype	Experimental	Army	
7	350	NEES-(RMI)	CML-2N	130	MA-MEA	172	Press.	—	12 Built	Gorgon P/A	Navy	Figs. 20, 21
8	500	GALCIT	—	20	RFNA-Gas.	185	Press.	350	Prototype	Experimental	Navy	
9	620 (220+400)	RMI	CML-5N	95	MA-MEA	190	Press.	—	280 in Prod.	Lark P/A	Navy	Fig. 22
10	700	RMI-(NEES)	CML-3N	65	MA-MEA	172	Press.	—	12 Built	Gorgon IIIc P/A	Navy	
11	710	Aerojet	XCNLT-600	297	NM	200	Turbopump	600	Prototype	Experimental	Navy	
12	750	Aerojet	—	—	—	—	—	—	—	—	—	
13	800	Goddard	A1	20	LIQ.O_2-Gas.	179	Turbopump	390	Prod.	Sounding Rocket Missile	—	
14	800	Goddard	—	30	LIQ.O_2-Gas.	—	Press.	—	1 Built	PBY JATO	Navy	
15	1000	Aerojet	25AL1000	25	RFNA-AN	159	Press.	—	60 Built	A-20 Aircraft Unit	Army	
16	1000	Aerojet	25ALD1000	25	RFNA-AN	154	Press.	—	200 Built	JATO	Army	Fig. 23
17	1000	GALCIT	—	20	RFNA-AN	160	Press.	—	1 Built	Experimental	Navy	
18	1000	GALCIT	Light Wt. Corp.	35	RFNA-AN	182	Press.	300	Prod.	Light Wt. Corp. Missile	Army	
19	1000	GALCIT	Corp. Model 6	90	RFNA-AN	182	Press.	300	Prod.	Corp. Model 6 Missile	Army	
20	1475	NEES	DU-1	35	RFNA-AN	150	Press.	—	4 Built	PBY JATO	Navy	Fig. 1
21	1500	Aerojet	38ALDW1500	38	MA-MEA	148	Press.	—	100 Built	JATO	Navy	Fig. 24
22	1500	GALCIT	GALCIT-139	47	RFNA-AN	—	Press.	—	Prod.	WAC Corp. Missile	Army	
23	2000	Aerojet	XCALR2000	Cont.	RFNA-AN	185	Pump	—	Prototype	Experimental	Army	
24	2000/250	Aerojet	XCALR2000	Cont.	RFNA-AN	—	Pump	—	5 Built	Experimental	Army	
25	2000	GALCIT	—	30	—	—	—	—	—	Experimental	Army	
26	2600	Aerojet	—	21	RFNA-AN-Alc.	—	Press.	—	10 Built	NIKE Missile	Army	
27	3000	Aerojet	X40ALD3000	40	RFNA-AN	154	Press.	250	16 Built	JATO	Army	Fig. 25
28	3100	RMI	M18-G2	52	LIQ.O_2-Gas.	179	Press.	—	Prototype	PBM Aircraft JATO	Navy	
29	6000	Aerojet	X35AL6000	38	RFNA-AN	149	Press.	—	Prototype	PB2Y-3 Aircraft JATO	Navy	
30	6000	Aerojet	X40AL6000	40	MA-MEA	149	Pump	—	Prototype	Experimental	Navy	
31	6000	RMI	1500-N4C	300	LIQ.O_2-Alc.+H_2O	190	Press.	—	6 Built	XS-1 Aircraft	A/N	Figs. 26, 27
32	6000	GALCIT	JL-563	25	RFNA-AN	—	Press.	300	Prototype	Experimental	Army	
33	20,000	GALCIT	Corp. 59	60	RFNA-AN	—	Press.	300	Prod.	Corp. 59 Missile	Army	

Table IV. Summary of German Rocket Engine Development.

No.	Thrust	Manufacturer	Designation	Duration in secs.	Propellants	Specific Impulse	Fuel Feed System	Chamber Pressure	Status	Use
1	308-66	BMW	109-548	22-17	NA-TONKA	156	Press.	400	1300 Built	X-4 Missile
2	660	Oberth-Von Braun	—	16	LIQ.O_2-Methanol	242	Press.	—	Prototype	A-1, A-2 Missiles
3	770	HWK	DFS-194	—	H_2O_2-$NaMnO_4$	106	Turbopump	270	Prototype	Experimental Aircraft
4	836-132	BMW	109-558	33-57	MA-TONKA	175	Press.	300	Prod.	Hs-117 Missile
5	836-132	HWK	—	60	MA-GAS.	156	Press.	400	Prod.	Hs-117 Missile
6	1100	HWK	RI-201/109-500	30	H_2O_2-$NaMnO_4$	100	Press.	270	150(+) Built	He-111 JATO
7	1320	HWK	RII-206/190-507B	10	H_2O_2-$NaMnO_4$	109	Press.	270	300 Built	Hs-293 Missile
8	1474	Schmidding	109-513	11	O_2-Methanol	175	Press.	—	Prototype	Hs-293 Missile
9	1650-440	HWK	RII-203	—	H_2O_2-$Ca(MnO_4)_2$	270	Turbopump	270	Prototype	Me-163A Aircraft
10	1760	HWK	LT-1200 and 1500	106	H_2O_2-Gas.-$NaMnO_4$	189	Press.	285	Prototype	Torpedo
11	2200	HWK	RI-203/109-501	42.5	H_2O_2-Gas.-$NaMnO_4$	172	Press.	300	Prod.	Aircraft JATO
12	2200/1320	Peenemunde	—	2-3	Visol	100	Press.	—	Experimental	Taifun Missile
13	2750/1320	BMW	109-718	112-180	NA-TONKA	180	Pump	525	Prod.	Me-262 JATO
14	3300	HWK	RI-209/109-502	30	H_2O_2-Gas.-$NaMnO_4$	180	Press.	320	Prod.	Aircraft JATO
15	3300	Army Weapon Dept.	—	45	LIQ.O_2-Methanol	143	Press.	—	30 Built	A-3, A-5 Missiles
16	3300-440	BMW	RII,-303, BMW-3390A	8-15 min.	NA-Methanol	175	Turbopump	500	30 Built	Me-163 Aircraft
17	3300-660	HWK	109-509A-1	251	H_2O_2-C.Stoff	181	Turbopump	300	300 Built	Me-163B Aircraft
18	3300-1540	HWK	RI-203/209	30-40	H_2O_2-B.Stoff-Gas.-$NaMnO_4$	156	Press.	300	4 Built	FV-143 Missile
19	3740-330	HWK	109-509A-2	64-120	H_2O_2-C.Stoff	181	Turbopump	300	2 Built	NATTER Aircraft
20	4400-880	HWK	109-509C	15-20 min.	H_2O_2-C.Stoff	181	Turbopump	320	Experimental	Me-163B-1 Aircraft
21	4400-2000	HWK	RI-210B	70	MA-GAS.	156	Turbopump	400	60 Built	ENZIAN Missile
22	4400-2200	Conrad	—	70	NA-Visol	181	Press.	300	Prototype	E-4 ENZIAN Missile
23	4800-3960	DVK	—	43	NA-Visol	181	Press.	300	260 Built	Rheintochter 3 Missile
24	5500-3300	Conrad	—	56	NA-Visol	181	Press.	300	Prototype	E-5 ENZIAN Missile
25	14,060	Conrad	F-55	7	NA-TONKA	169	Press.	300	3 Built	Feuerlilie Missile
26	17,700	Peenemunde	—	42	MA-Diesel Oil	180-195	Press.	—	300 Built	Wasserfal Missile
27	55,000	Peenemunde	—	68	LIQ.O_2-Alc.	200	Turbopump	300	Prod.	A-4 Missile
28	440,000	Peenemunde	—	50	LIQ.O_2-Alc.	—	Tubopump	—	Design Study	A-10 Missile

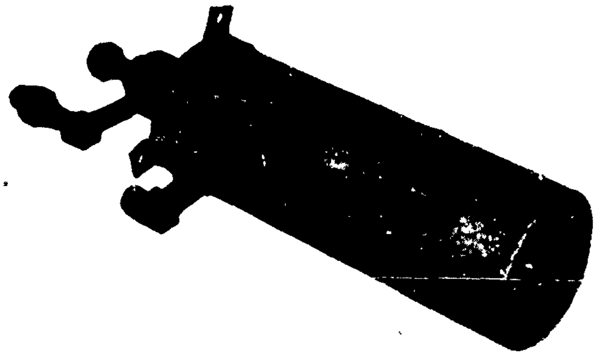

Figure 20. Reaction Motors, Inc., CML-2N "Gorgon" rocket motor.

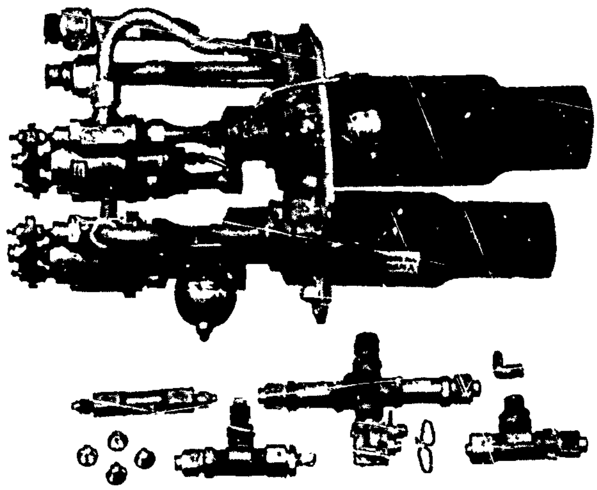

Figure 22. Reaction Motors, Inc., CML-5N "Lark" rocket motors.

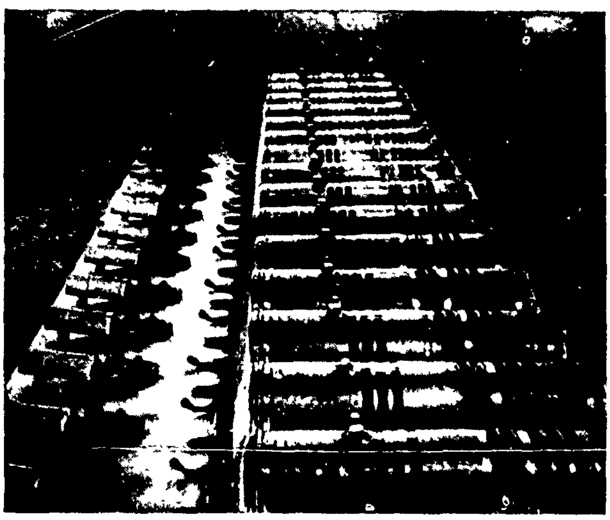

Figure 21. Reaction Motors, Inc., CML-2N production.

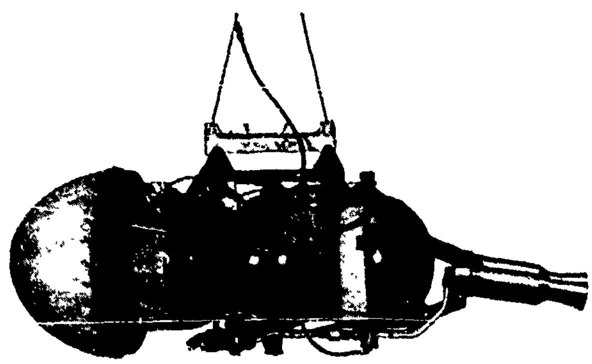

Figure 23. Aerojet 25ALD1000 jet assisted takeoff unit.

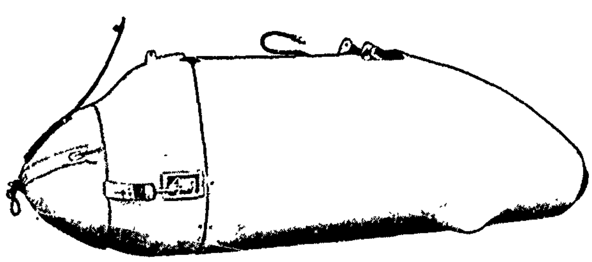

Figure 24. Aerojet 38ALDW1500 jet assisted takeoff unit.

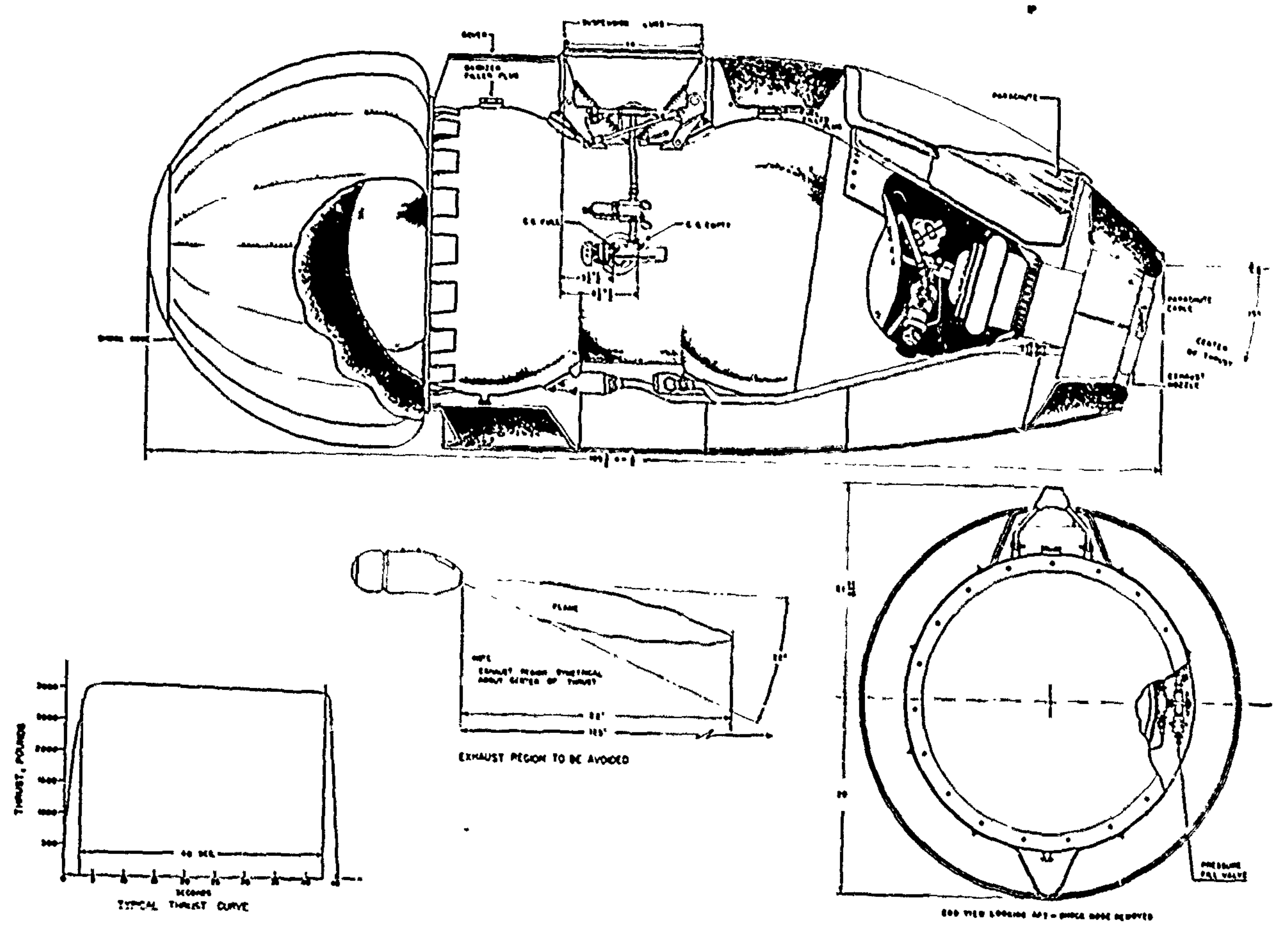

Figure 25. Aerojet X40ALD3000 jet assisted takeoff unit.

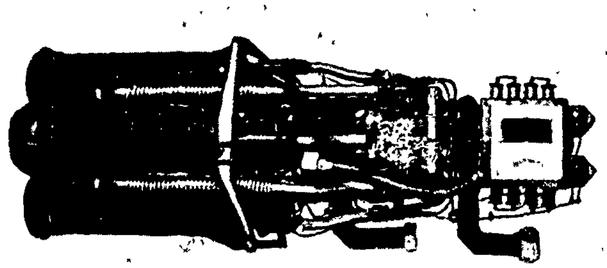

Figure 26. Reaction Motors, Inc., 1500N4C XS-1 rocket power plant.

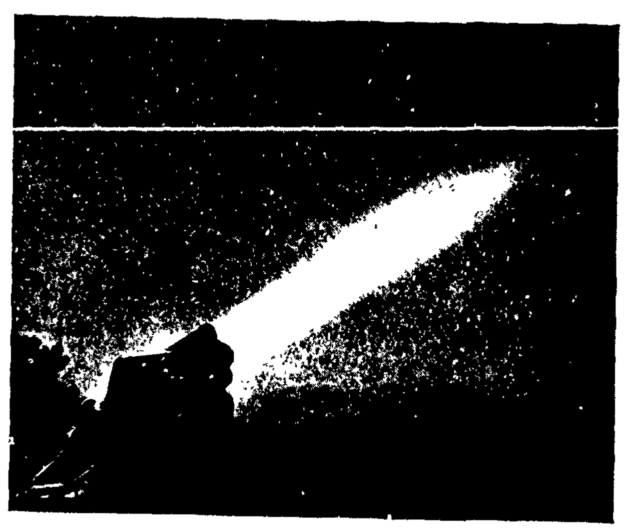

Figure 27. Reaction Motors, Inc., 1500N4C test stand operation.

V. REVIEW OF CURRENT LIQUID ROCKET ENGINEERING RESEARCH AND DEVELOPMENT PROBLEMS

Approximately 75% of the current development work in liquid rockets can be conveniently divided into four propellant systems. These are:

1. Acid oxidizer systems.
2. Gaseous and liquid oxygen oxidizer systems.
3. Hydrogen peroxide systems.
4. Nitromethane systems.

The work of the various agencies in each of the above categories is taken up alphabetically and outlined below. Three additional sections are included to cover the remaining problems. These are:

5. General propellant studies, including high energy fuels and systems.
6. Rocket motor cooling, combustion chamber, and injector studies.
7. Component development and miscellaneous liquid rocket problems.

Table V summarizes the problems directed toward the design or development of a specific power plant. The abbreviations on page 23 are used here. Technical progress, in general, is covered up to March 1, 1947. In a few cases the data are more recent and others somewhat earlier.

A. Nitric Acid Oxidizer Systems and Components

1. AEROJET ENGINEERING CORPORATION.

a. A 1,000-lb. thrust acid-aniline unit for a ground to air missile. The preliminary design is complete and the unit will be a lightweight pressurized system with diaphragm starting. Boeing Aircraft is the prime contractor.

b. A 2,600-lb. thrust red fuming nitric acid-aniline and furfuryl alcohol pressurized system for the Aerobee XASR-1 Sounding Rocket. The system incorporates diaphragm starting and has a design duration of 45 seconds. The exhaust nozzle has an area ratio of 5.5, giving full expansion at 15,000 ft. and a maximum or vacuum thrust of 3,100 lbs. The combustion chamber pressure is 300 p.s.i. for 2,600-lb. sealevel thrust. The motor, complete with burst diaphragm holders and filters, weighs 30 lbs. Specific impulse is 193 lb. sec./lb. Twenty units total are required.

The prototype design is approximately 90% complete. Preliminary tests were run on a 1,000-lb. motor fired vertically and using diaphragm starting. Prototype tests are completed, and the unit is in production. This work is being done under Bureau of Ordnance Contract NOrd-9837.

c. A 2,600-lb. thrust red-fuming nitric acid and aniline furfuryl alcohol system complete with valves and regulators is being produced for the NIKE missile. Ten units are required and nine have been delivered. The motors weigh 38 lbs. and deliver 2600-lb. thrust at sea-level at a specific impulse between 186 and 193 lb. sec./lb. for 21 seconds. A program for work in the next fiscal year calls for 24 lightweight motors and systems using diaphragm starting. The prime contract for this work is held by Bell Telephone Laboratories with Army Ordnance. Aerojet holds a subcontract with Douglas Aircraft who in turn holds a subcontract with Bell Telephone Laboratories.

d. A 4,000-lb. thrust 60-second red fuming nitric acid and 80% aniline, 20% furfuryl alcohol pressurized droppable JATO unit is being developed for the XB-45 airplane. This unit is designated 60ALD4000. The motor weighs 15 lbs. and valves 8 lbs. Specific impulse is 186-193 lb. sec./lb. All components are ready for test, but two prototype motors have been redesigned for test due to difficulties encountered in cooling at shutoff. Fifteen units are on order and an additional 12 sets of spare motor and valve assemblies will be required. This work is Phase 1 of Army Air Forces Contract No. W-33-038-ac-14549.

e. A 4,000-lb. thrust red fuming nitric acid and 80% aniline, 20% furfuryl alcohol JATO unit is under development for the XB-47 airplane. The motor and valve assembly will be the same as for the 60ALD4000 unit above. However, in this case the unit will be a fixed installation. Consideration has been given to driving the turbine for the propellant pumps from the turbojet air supply. This has not been definitely established and may turn out to be impractical. Design studies are underway. This work is Phase 2 of Army Air Forces Contract W-33-038-ac-14549.

f. A 40,000-lb. thrust 90-second turbopump-fed rocket engine, tentatively using white nitric acid and furfuryl alcohol as propellants, is under development. The problem is purely experimental and has no specific application. The design study is complete and includes possible variation in thrust down to 10,000 lbs. by variation of feed pressure. A 20,000-lb. motor will be built to run initial tests and gain information on the problem. This motor will be water-cooled for the first tests but eventually will be cooled with the fuel, if practical. The choice of 20,000 lbs. for the initial test motor was influenced by the fact that it is felt that 20,000 lbs. is the largest size motor that may be safely run at the Azusa plant.

A screw pump design similar to that of the ME-163 Walter booster pump is being investigated as the first stage of a two-stage fuel pump. Tests will be run with an electric motor drive to study the feasibility of this design and with particular reference to minimum tank pressures to prevent cavitation. The two-stage pump is proposed with the motor cooling jacket between stages to reduce the compression load on the combustion chamber.

The preliminary design study of this project is included in Aerojet Report No. 219. Army Air Forces Contract W-33-038-ac-15309 covers this development.

2. Bendix Aviation Corporation, Eclipse Pioneer Division.

a. A pumping system for a 620-lb. thrust mixed acid and monoethylaniline rocket, designated the CMLAN-1 is being developed. Reaction Motors, Inc., as a subcontractor is to supply the 220-lb. and 400-lb. thrust motors, propellant valves, and test facilities. The development of this engine is essentially the development of the pumping system, and it is about 75% complete. The system is required to deliver a 220-lb. continuous thrust with a 400-lb. intermittent thrust when required. This unit will be required to operate for three minutes. The weight including pumping system is approximately 35 lbs. The work is being carried out under Bureau of Aeronautics Contracts NOa(s) 8060 and NOa(s) 8396.

3. Jet Propulsion Laboratory, California Institute of Technology.

a. A program is under way to obtain data on motors using furfuryl alcohol with white fuming nitric acid. This combination possesses certain handling advantages over the red fuming nitric acid-aniline combination. The considerable decrease in toxicity may compensate for a slight loss in specific impulse.

b. An extensive program is planned on the development of acid-type motors operating at high chamber pressures. For turborocket systems it appears that the combined weight of propellant and propulsion system can be reduced by operation at chamber pressure above those in common use today. The program initially calls for studies of performance, heat transfer, etc. at chamber pressures up to about 1,000 p.s.i. The ultimate aim is the development of a high pressure regeneratively cooled motor.

c. Experimental studies are planned on the possibility of producing a high thrust, short duration, acid-type rocket, particularly for ordnance applications. The chief problem will be the attainment of extremely short starting periods.

d. Highest priority is being given to experimental studies on a 200-lb. scale of nitric acid-hydrazine motors. The test work is being carefully planned since only limited quantities of the very expensive hydrazine are available. It is hoped to obtain sufficient data to guide the design of a 2000-lb. thrust regeneratively-cooled motor.

e. Among contemplated programs are included studies on nitric acid-ammonia units and systematic tests of the influences of propellant temperature on performance. Equipment for the latter work is on hand and actual test work should be forthcoming very shortly.

f. The JPL has also carried through the design and development of the nitric acid-aniline motor used in the WAC Corporal. Refinements on the design have been made and considerably higher performance is expected in future tests. The design of the 20,000-lb. thrust propulsion unit for the Corporal, a 10,000- to 12,000-lb. test vehicle with a range of 50 to 85 miles, has occupied the laboratory for some time. Two configurations are involved. The first, designated as Corporal E, is to use a pressure feed system. The second design, designated as Corporal F, utilizes a turbopump system. This work is being done under the sponsorship of Army Ordnance.

4. M. W. Kellogg Company.

a. A 6,000-lb. thrust rocket motor is being developed. This rocket will be the result of a survey program on possible rocket propellants for aircraft use. The study is approximately three-fourths complete. The conventional fuels (carbon, hydrogen, nitrogen, and oxygen compounds) are about finished, and it is planned to do some analytical work with boron and halogen flouride compounds. Results so far indicate the choice of white fuming nitric acid and anhydrous hydrazine.

One thousand pounds of anhydrous hydrazine were purchased from the Fairmount Chemical Company. Funds remaining for this project will not permit the purchase of more of this propellant, but the AAF has indicated its intention to purify and concentrate the material seized from the Germans and to furnish Kellogg with about three or four tons of the anhydrous material.

A 50-lb. thrust motor has been run to check the calculated values of the survey. The principal fuels tested have been: anhydrous hydrazine, hydrazine monohydrate, the low freezing point eutectic mixture of hydrazine and 23% water, and furfuryl alcohol.

White fuming nitric acid was mainly used as the oxidizer; some work with 90% peroxide was done. Theoretical values and GALCIT experimental values were closely checked. The hydrazine fuels ignited and burned well with a short nonluminous flame. It was found that the same combustion and injection methods could be used for anhydrous hydrazine and for the eutectic mixture, and this suggests a way of getting around the freezing difficulties. Impinging jets gave satisfactory results.

Additional tests with white fuming nitric acid and hydrazine have been run in a 500-lb. 10-second uncooled motor. Experimental and calculated values check well and results also are in accord with the 50-lb. tests. A few acid-aniline runs were also made.

Injection has not been thoroughly studied. So far a modified GALCIT design of impinging stream injector has been used. Other injectors such as cone-splash plate and spray have been designed and will be tested shortly.

Future plans are to design, fabricate, and test a 1,500-lb. regeneratively cooled motor. Upon satisfactory completion of this step the same will be done for a 6,000-lb. motor. Experimental work is due to be completed by September 1, 1947, and the final report two months later.

This work is being done under Army Air Forces Contract W-33-038-ac-13916, restricted classification.

b. A 14,000-lb. thrust rocket unit is being developed to consist of two chambers, one giving a 10,000 lb. and the other a 4,000-lb. thrust. Each motor must be capable of variation of thrust down to 40% and also capable of independent or simultaneous operation. A hydrocarbon and nitric acid are specified as propellants.

This contract has only been in effect since February 1, 1947; consequently, little work has been done. Gasoline and white fuming nitric acid have been tested in a 50-lb. motor. Ignition was achieved by a slug of furfuryl alcohol or hydrazine in the gasoline feed line. It is planned to try to find additives to make the fuel self-igniting.

This work is being done under Army Air Forces Contract W-33-038-ac-15313, confidential classification.

5. Naval Air Missiles Test Center, Pt. Mugu, California.

a. TED PAU-PP-204. This assignment covers the development of the "Lark" CML4N turbopumping unit which develops a 620-lb. thrust maximum with a 220-lb. and a 400-lb. motor. The turbine is driven by the exhaust of the 220-lb. cruising motor (Fig. 17). The 400-lb. motor is cut in and out, depending upon the flight requirements.

Six self-pumping runs have been made on the test setup. Starts and stops are smooth, but a low frequency chug has developed which will have to be removed before it may be considered satisfactory. New turbines are installed for each run, the motor is started by the 30 p.s.i. tank pressurizing air. An auxiliary blast of air on the turbine is used to bring the pressure up to about 250 p.s.i. where the jet picks up and takes over. There appears to be some tendency for the system to stabilize at a low chamber pressure of around 180 p.s.i. Operation of the turbine blades in the wrong portion of the lift curve during starting, and acceleration may be responsible for this. An investigation of this is planned. In general there are no serious problems ahead in the development, and in a short time a satisfactory unit is expected. NAMTC weekly and monthly progress reports to BuAer describe the status of the problem.

A stand-by crew is available for flight testing of the pressurized "Lark" units at NOTS Inyokern. Some difficulty has been experienced in filling the propellant tanks with the slosh-preventing bladders, but the problem does not appear to be too serious. Two successful flight tests have been made with the KAQ. Ranges of about 18 miles and altitudes of about 25,000 ft. have been obtained.

b. A project (TED-PAU-PP-206) has been set up for the testing of captured German equipment. Three of these units use nitric acid as the oxidizer. The test stand and units are being prepared for test, but no tests have been run to date. Lack of personnel has delayed this work. The tests are being run primarily for educational and familiarization purposes.

1. BMW 3395/SK JATO unit for ME-262. This is a 2,750-lb. thrust unit utilizing nitric acid and 50% triethylamine and 50% commercial xylidines as propellants. This is a pump-fed system, deriving its power from the BMW turbojet on which it was mounted. Reference 29 contains an excellent description of this rocket power plant.

2. X-4 anti-aircraft missile power plant. This unit utilizes nitric acid and aniline as propellants and has tubular spiral tanks from which the propellants are driven by free pistons forced by a pressurizing gas. The motor develops a thrust of about 200 lbs. for a short duration, 10 seconds or under.

3. HS117, ground to air missile power plant. This rocket power plant utilizes nitric acid as oxidizer and 57% crude m-xylidine, 43% triethylamine as fuel. The thrust is approximately 485 lbs. for 57 seconds. The thrust of this motor is variable by means of cut-

ting out pairs of injector holes, thus holding the mixture ratio constant while cutting down the fuel flow.

c. TED-PAU-PP-205 is a project for the theoretical and experimental investigation of a high altitude sounding rocket. The original design calls for a 6,000-lb. thrust white acid-furfuryl alcohol unit with a 93-second duration. A turbine driven by the exhausts of the four component motors drives the two propellant pumps. The pump and turbine for this setup has been designed and built by James Coolidge Carter, pump consultant, of Pasadena, California (Fig. 15). The pump turbine combination as delivered weighs 4.5 lbs. The tests on this pump will be run as soon as the new pump test stand arrives. Detail design of the remaining components has been hampered by lack of qualified personnel.

6. Reaction Motors, Incorporated.

a. The design development and testing of an igniter for acid-gasoline rocket motors. This work is complete[4] and was conducted under Task Order VI BuAer Contract NOa(s) 7866.

b. The testing and evaluation of iron-free mixed acid as an oxidizer in the CML5N-1 220-lb. rocket motor, through performance comparison with the mixed acid now used as an oxidizer. Numerous tests were run to provide a sound basis of comparison including an evaluation of starting characteristics. This work is complete.[5] This work was conducted under Task Order XIII of BuAer Contract NOa(s) 7866.

c. The design, development and fabrication of propellant and pressurizing gas, portable servicing equipment to be used primarily in conjunction with the "Lark" and "Gorgon" guided missile testing programs. This equipment includes one acid servicing unit mounted on a Mark II Bomb Cart, incorporating a hand-driven pump, two 55-gallon acid drums and suitable transfer hoses, a similar aniline servicing unit, and a pressurization unit mounted on a Mark III Bomb Cart. The latter unit includes a 12.2 cu.ft./min. 3,000 p.s.i. air compressor, and suitable storage and transfer equipment.

This equipment is practically complete and six sets of units are required under BuAer Contract NOa(s) 8358.

d. An experimental investigation is being conducted to determine the performance characteristics of furfuryl alcohol and other fuels when used in combination with nitric acid as an oxidizer. The work includes the variation in specific-impulse with propellant mixture ratio and combustion chamber pressure. It also includes an investigation of white nitric acid, red fuming nitric acid and mixed acid when used with furfuryl alcohol. Progress on this problem is reported in progress reports to the Bureau of Aeronautics under Contract NOa(s) 8540 Task Order VII.

e. The design, development, construction, and testing of a 2,000-lb. variable thrust nitric acid-monoethylaniline rocket motor. Variable thrust is obtained by varying the flow rate of the propellants and by the incorporation of a variable area nozzle involving the use of a restrictor. This restrictor may be inserted or withdrawn from the nozzle throat to vary the area. The oxidizer is used to cool the motor jacket and the fuel to cool the restrictor. The unit has been designed to use a pressurized propellant feed system.

A standard motor from the A6000C4 liquid oxygen power plant has been modified to run on the specified propellants, and satisfactory preliminary runs have been made without the restrictor. The design is complete, however, and the parts are being fabricated. This work is being done under Task Order XI of BuAer Contract NOa(s) 8540.

B. *Gaseous and Liquid Oxygen Oxidizer Rocket Systems and Components*

1. Aerojet Engineering Corporation.

a. The original work at Aerojet on gaseous hydrogen and oxygen motors was done on BuAer Contract Noa(s) 7968 Task Order No. 6. The two objectives of the task were: to survey design criteria and material requirements for high performance rocket motors, and to design, build, and test rocket motors using gaseous hydrogen and oxygen as propellants with the goal of a 100 to 400-lb. thrust motor capable of operation for one minute without damage while delivering a specific impulse of 300 seconds.

The following tests were conducted in performance of the contract (6):

1. A 165-lb. thrust porous bronze motor cooled by water transpiration was run for a total of 105 seconds. Jet velocity was 9,330 ft./sec., $I_{sp} = 291$.

2. A 100-lb. thrust motor with one part at a time cooled by water transpiration was run for a total of 371 seconds. Performance was unsatisfactory, due to excess coolant requirements.

3. A 100-lb. thrust porous ceramic motor cooled by the propellant gases was run for a total time of 72 seconds, but it failed on each run due to lack of strength of the ceramic.

[4] "Progress Report on Acid-Gasoline Igniter," Reaction Motors, Inc., dated 15 May 1946 and Development of a Compressed Air-Gasoline Igniter," Reaction Motors, Inc., dated 22 August 1946.

[5] "Testing and Evaluation of Iron Free Acid," Reaction Motors, Inc., Report No. 168N-1, dated 2 October 1946. *Restricted.*

4. A 100-lb. thrust motor with porous bronze gas transpiration cooled chamber was run satisfactorily for 20-second periods. The performance was 11,490 ft./sec. and I_{sp} = 357 seconds at P_c = 289 p.s.i.a. and H_2/O_2 molar ratio of 8:1.

5. A 400-lb. screen grid motor cooled by the propellant gases failed due to mechanical stresses.

6. 100-lb. thrust motors with waste water jacket cooling were run for a total time of 18 minutes. At P_c = 500 p.s.i.a. and H_2/O_2 molar ratio 3:1, the jet velocity was 10,850 ft./sec. at an I_{sp} of 336 seconds.

7. Radiation cooling was considered but rejected on the basis of materials available at present.

8. 180-lb. thrust copper heat capacity motors were run for a total time of 72 seconds at not longer than 15 seconds per run. Jet velocities of 11,300 ft./sec. were reached at a chamber pressure of 275 p.s.i.a. and H_2/O_2 molar ratio of 4:1. I_{sp} = 352 seconds.

9. Flared tube heat capacity type motors of L* from 5 to 7 inches have shown specific impulses up to 352 seconds.

10. Flared tube motors with an L* of 3 and water transpiration cooling gave a net I_{sp} of 291.

b. BuAer Contract NOa(s) 8496 has been let for the continuation of the development of hydrogen and oxygen rocket motors. This work has been broken into two phases:

1. Design development and testing of a 1,000-lb. thrust gaseous H_2/O_2 motors including studies of small L* motors with improved cooling methods.

2. Design study of a 300,000-lb. liquid H_2 and O_2 turborocket power plant.

Results of tests under (1) above thus far completed indicate that for flared tube motors of the heat capacity type the optimum L* is 6 in. to 8 in. and the I_{sp} = 354. A flared tube motor of the waste water-cooled type at L* = 5.5 in. the motor gives an I_{sp} = 334; for a tubular motor the optimum L* lies between 7 in. and 10 in. and gives an I_{sp} of 284 seconds, for a 4.25:1 H_2/O_2 molar ratio. A flared tube motor transpiration cooled with water at L* of 5.75 gave an I_{sp} of 258.

Tests with tubular type motors with and without flare indicate a high pressure gradient along the chamber. The pressure at the injector does not correspond to chamber pressures in a conventional motor. The ratio of the exit or throat pressure to the initial pressure is constant for a motor length and is independent of initial pressure. The exit pressure of a tubular motor is between 30% to 40% of the initial pressure, and varies inversely with L*.

The heat transfer measured in a flared tube motor has an average value of about 6 Btu./in.²/sec. and a peak value of 15 Btu./in.²/sec. at the throat.

The following permeable materials are now available commercially: graphite, bronze, silicon carbide, nickel, stainless steel, tungsten, and molybdenum. Parts 1½ in. in diameter and up to 4½ in. long may be made from the available sizes. The high pressure gradient in tubular and flared tube motors complicates the transpiration or sweat cooling problem. Presently available materials have too high a permeability.

The design study of the 300,000-lb. thrust motor has been completed, resulting in a single chamber weighing approximately 2,675 lbs. Radiation cooling has been abandoned because of the increase in weight required. The turbopump will be required to develop 8,400 h.p. and will weigh 850 lbs. Valve and plumbing design have led to a weight estimate of 355 lbs. for these components.[6]

2. Bell Aircraft Corporation.

a. All work conducted thus far at Bell has been on liquid oxygen and alcohol or gasoline, and it is felt that with the limited number of people available it would be inadvisable to branch into other fields.

Various size motors from 250-lb. to 10,000-lb. thrust have been run. Bell Aircraft is working on Project METEOR under Bureau of Ordnance Contract NOrd. 9876. An outline of the proposed program as received from Bell is given below (24).

1. *Combustion.* The effects of chamber size and shape, and injector design on combustion will be investigated both theoretically and experimentally. The relation of chamber geometry to injector design is presently being investigated in 250-lb. thrust motors. Chamber *L/D* and *L** are also being investigated for various mixture ratios and injectors. To date six types of injectors have been investigated, and four more are under construction.

2. *Performance.* Optimum performance of rocket missiles involves the proper choice of propellant, propellant pumping system, type of rocket motor cooling, nozzle expansion ratio, and optimum flight path. The successful integration of these factors involves extensive studies. Calculations have been completed of the physical characteristics of the combustion gases for the entire range of mixture ratios for the chosen propellants.

3. *Rocket Motor Cooling.* The program attacking the regenerative cooling of rocket motors includes: metallurgic and structural analysis; establishment of practical liquid film coefficients; and establishment of

[6]References 2, 6, 8, and 10 give a detailed account of this work. A portion of the above material was taken from these sources.

coefficients representing the cumulative effect of the combustion gas temperature, the gas film coefficient, and the shape of the rocket nozzle. Combustion chamber wall and coolant temperatures are being measured at frequent intervals in 250-lb. and 1,500-lb. rocket motors, using water as a coolant. Bell is also manufacturing and testing rocket motors which are being lined or coated (by Massachusetts Institute of Technology) with ceramic materials.

4. *Rocket Motor Design or Construction.* This is an attempt to adapt the processes of spinning, stamping, machining, electro-forming, etc. to inexpensive production of rocket motors.

5. *Propellant Supply Systems.* Design studies will be made to determine the optimum propellant supply system for any application. A turbine pump unit has been designed, constructed, and is now being tested. The General Electric Company has been cooperating in this work. Control systems for the pump and motor are receiving considerable attention. A pressurized fuel feed system has also been constructed and tested.

6. *Valves, Controls, and Hydraulic Components.* This program covers the study and development of the following items: igniter, check valves, regulator valves, electric control valves, frangible disc valves, fittings and gaskets. Some work has been done on frangible disc valves, regulator valves, and solenoid control valves.

b. A 1,500-lb. thrust pressurized alcohol-oxygen system is being designed for a Project METEOR test vehicle.

3. CONSOLIDATED VULTEE AIRCRAFT CORPORATION.

a. Tests are being conducted on a ducted rocket or a ram jet using a small rocket motor as the ignitor and flame stabilizer for the ram jet fuel. This work is being done under an AAF contract for the XP-92 airplane. A Reaction Motors A50C1 engine (50-lb. thrust liquid oxygen-alcohol rocket motor) is being used for these tests. The tests are intended to determine the mixing length required to complete the mixing of the rocket jet and air stream and to determine the burning characteristics of gasoline injected into the air stream under conditions of complete mixing. Successful tests have been made with a duct 13 in. in diameter and 12 ft. long with a 3:1 diffuser ratio under simulated flight speeds of 600 m.p.h. The gasoline was injected through a group of perforated rings concentric about the rocket nozzle. Even burning was obtained at all fuel flows. The maximum fuel flow was limited to a gas temperature of 2,500°F because of the thermocouple installation and there appeared to be no lean limit to combustion. Testing with the present equipment will continue with respect to the methods of fuel injection, mixing characteristics, combustion efficiency, thrust measurements, temperature studies, and optimum dimensions of complete engine. Later it is planned to use the Roots blowers as exhausters for altitude testing.

4. CURTISS-WRIGHT CORPORATION, PROPELLER DIVISION.

a. A power plant for Bell XS-2 airplane is being built under AAF Contract W-33-038-ac-16269. The thrust requires one 10,000-lb. motor and one 5,000-lb. motor. Liquid oxygen is the oxidizer, with a 75% alcohol, 25% water mixture for fuel and coolant. Delivery requirements call for four units plus one prototype. It is expected that the prototype will be ready six months after test facilities are available.

The system will be complete with turbopump unit and will be continuously throttleable from 2,500 lbs. to 15,000 lbs. It is planned to pressurize the oxygen tank with liquid oxygen drawn from the feed line and passed through a heat exchanger for gasification and thence back to the tank. The fuel will be pressurized from steam taken from the second stage of the turbine. Approximately 30 p.s.i. will be used on all tanks.

The coolant portion of the fuel first passes through a cooling jacket around the nozzle and then is injected tangentially to the chamber wall at the head of the combustion chamber to provide film cooling in this region.

All work is in the design stages. The 15,000-lb. thrust system two-stage turbine and pump system layout is about complete. The estimated weight is 70 lbs. It is expected that the system will operate at an I_{sp} of better than 185 at sea level.

b. AAF Contract W-33-038-ac-14827 was let to Curtiss-Wright on an educational and developmental basis. Three phases were assigned:

Phase 1. A 10,000-lb. thrust motor and a 15,000-lb. capacity turbine-driven pump with combustion pot and controls. This system is to be designed for a hydrocarbon fuel, a coolant, and liquid oxygen. The fuel will probably be JP-1 or a new type JP fuel. Chamber pressure will be approximately 400 p.s.i. and the system is to be designed for 120 seconds running time. The pumping system will apply in part to the XS-2 contract and to the XP-91, but it was not assigned as such. The design of certain components is nearly complete, but no construction has been started at this date. The project is expected to be completed in early 1948.

Phase 2. Design one 60,000-lb. thrust motor and system with turbopump-fed propellants. The propellants will be liquid oxygen, a coolant, and a hydrocarbon fuel. A separate combustion pot for the turbine will be used. No particular application is in mind for this power plant, but it may be considered for missile application. No work has been started to date, and the actual starting time will be greatly influenced by the availability of facilities.

Phase 3. Design study and development of a large power plant, the size of which has not been established, but will probably be over 100,000 lbs. No work has been started.

c. AAF Contract W-33-038-ac-14171 requires the design, prototype, and construction of two droppable ATO units of 4,000-lb. thrust and 60-second duration with 10% excess tankage. The unit is not to be expendable and is intended for use on the XB-45 airplane. The unit is to be complete with tanks, turbine-driven pumps and parachute. The propellants will be liquid oxygen and gasoline with a mixture of water and alcohol as coolant. A motor has been built that can be used, but it is felt that a better and lighter motor could be built if the contract can be revised to cover this. This equipment will probably be completed in the fall of 1947. The turbine pumping system has been designed and parts made. The turbine unit will be tested, using steam from a commercial type boiler.

d. Two additional projects have been proposed but no work is under way pending formal contracts.

1. A 16,000-lb. thrust JATO unit has been proposed for the XB-47 airplane using liquid oxygen and kerosene as propellants with alcohol and water as coolants. Either two or four motors are proposed in a fixed installation using a turbine-driven pump which derives its starting air from the turbojet compressor.

2. A JATO, climb and cruising power plant has been proposed for the XP-91 airplane where the main source of power is a 5,000-lb. thrust turbojet. The JATO and climb unit would consist of a 16,000-lb. thrust motor, using a liquid oxygen and a JP type fuel as propellant with alcohol water coolant. This system will be divided into four 4,000-lb. thrust motors. Turbopumps will supply the propellants for 2.5 minutes and tanks will be jettisonable. A small 1,200-lb. thrust system consisting of two 600-lb. thrust motors will be used for cruising for five minutes. This system will use the same propellants but will be completely separate from the JATO unit, including propellant pump and tanks. Four production power plants and one prototype will be required.

5. Jet Propulsion Laboratory, California Institute of Technology.

a. A series of studies are being made on motors of the oxygen-ethanol and oxygen-methanol types to investigate such items as heat transfer, injector design, combustion volume, etc. This work is carried out on the 200-lb. thrust scale. This size of motor is favored by the JPL for work of this type, as it appears to be large enough so that extrapolation to large motors can be made successfully and yet small enough to be on a laboratory scale.

b. Plans are under way for a cooperative scheme leading to the construction of a plant for the production of liquid hydrogen. Liquid oxygen-liquid hydrogen motors, not exceeding a 200-lb. thrust size, are to be studied.

c. It is felt that hydrazine is one of the most promising high performance fuels available. Unfortunately its present high cost and difficulties of procurement have allowed only a limited number of small-scale tests on a 50-lb. scale, using the liquid oxygen hydrazine combination.

d. Initial tests have been run on the 50-lb. scale, using liquid oxygen and ammonia. An intensive program is under way.

e. Methyl amine as a fuel is presently under investigation by the chemistry section. As soon as possible, it is planned to test liquid oxygen-methyl amine motors on a small scale.

Items b through e, above, are covered in more detail in Volume I, Part 2 on *Fuels* of this Survey.

6. General Electric Company, Project HERMES, Malta Test Station.

a. The present primary objective of this group is the construction of a 16,000-lb. thrust liquid oxygen alcohol rocket motor of 50 seconds duration as a power plant for HERMES A-1. This work is under the sponsorship of Army Ordnance. The work at the Malta test station is aimed at establishing the necessary techniques and design parameters on reasonable scale motors so as to be able eventually to construct the 16,000-lb. motor.

Tests are being conducted on a 1,300-lb. thrust liquid oxygen and alcohol-water motor. The G.E. group feels that combustion chamber length rather than L* is the critical design parameter for motors and have established 13½ in. or less as the optimum length for this size of motor. A specific impulse of 225 seconds has been demonstrated at chamber pressures of around 350 p.s.i. and at a mixture ratio of oxygen to fuel of 1.5 to 2.0. The motors have been water-cooled for these tests and the nozzle and combustion chamber cooling jackets are separate so as to determine the relative heat flow in each section. The coolant flow in the nozzle follows a labyrinth path back and forth in the direction of the axis of the motor. A helical path is used for the coolant flow around the combustion chamber. The injector design consists of a series of concentric rings of small holes alternating in fuel and oxidizer. These have worked satisfactorily, although mixing is improved when the holes are changed to slots.

Similar motors of 4,000-lb. thrust are run on another test stand for periods up to 25 seconds. It is planned to increase the size of the propellant tanks so that either longer runs may be made or larger motors up to 10,000 lbs. may be tested on this stand. Careful planning and checking of instrumentation is carried out on all work to insure accuracy of results. The flow rate of the alcohol is checked by three different methods during a single run. The liquid oxygen flow is measured by continuous weighing during the run and then corrected for the weight of pressurizing gas.

b. Gaseous oxygen and hydrogen have been run in a 50-lb. motor to gain experience in the operation of motors at high temperatures. This was done in preparing for testing liquid oxygen diborane motors. A few qualitative runs were made on a small motor with the latter propellants and it was found that an oxide coating formed in the motor which seemed to materially reduce the critical cooling problem. It was found necessary, however, to redesign the injector and allow a bleed of nitrogen gas to prevent the forming oxide from choking the injector. More elaborate quantitative tests are planned for the near future.

7. M. W. KELLOGG COMPANY.

a. A design study is being conducted under a subcontract with Republic Aviation. The prime contract W-33-038-ac-14208 is for an AAF guided missile. The propellants being considered for this problem are liquid oxygen and anhydrous hydrazine. The missile is planned to have two steps: the first one being 290,000-lb. thrust for 82 seconds, and the second a 70,000-lb. thrust for 82 seconds. Turbopumps will be used for both steps.

8. NORTH AMERICAN AVIATION, INC.

a. The primary concern of the Propulsion Development Section of the Aerophysics Laboratory is the development of a suitable rocket power plant for a long-range missile along the lines of the German A-4b (V-2 with wings). The general plan of attack is to use as a basis the liquid oxygen-alcohol propulsion system of the V-2, making such improvements and changes as time and testing facilities allow. Plans call for a continual development program including the possibility of eventually switching to a higher performance propellant combination. Motor chambers of 300-lb. and 3,000-lb. size are being investigated from the point of view of geometric configurations, injector design, cooling requirements, etc. Turbopump, gas generator, valve, and control system developments are in the early stages. A transpiration or sweat cooling investigation is planned for the near future.

The secondary concern of the Development Section is the development of a complete rocket power plant for the NATIV missile. The motor is to be furnished by Aerojet. It is a 2,600-lb. thrust acid-aniline unit. The system incorporates air pressurized propellant tanks which are an integral part of the missile air-frame. The unit will have a 31-second duration. The complete system will be static tested at the isolated Field Laboratory with a special test stand constructed for the purpose.

Jet vanes for the NATIV are being tested aerodynamically in the small supersonic wind tunnel and life tests of the various vane materials are being made, using an Aerojet 38-ALDW-1500 JATO unit.

This work is being carried out under AAF Contract W-33-038-ac-14191.

9. OHIO STATE UNIVERSITY.

a. The work thus far at OSU has been the development of manufacturing techniques of liquid hydrogen, along with detailed research on its chemical and physical properties and its effect on contacting materials. This work is covered in detail in Volume I, Part 2 of this Survey, entitled *Fuels*.

Investigations of pump design for pumping both liquid oxygen and liquid hydrogen are in progress, and liquid hydrogen has been successfully pumped. An uncooled liquid hydrogen-liquid oxygen rocket motor designed to develop a 100-lb. thrust at 300 p.s.i. chamber pressure has been constructed and tested with successful results. Vacuum jacketed fuel tanks and fuel lines and pressure controlled fuel valves for use with liquid hydrogen have been operated satisfactorily. The present work has been carried out under AAF Contract W-33-038-ac-14794.

Ohio State is now negotiating a contract with the AAF for complete construction and testing of a 1,000-lb. thrust liquid hydrogen-oxygen rocket power plant.

10. REACTION MOTORS, INC.

a. A series of tests were run to prove the efficiency and reliability of AN-18-1, AN-A-24, and type C Navy Department Specification No. 51-A-13a alcohol (versus Synasol) as a fuel for the 6,000-lb. thrust liquid oxygen-water alcohol rocket power plant. These are standard alcohols used by the Navy, and tests were made with various water-alcohol mixtures and with different fuel oxidizer ratios. This work was done under Task 8 of BuAer Contract NOa(s) 7866 (48).

b. The development, construction, and testing of a liquid propellant rocket engine to drive the rotor of a rotary wing aircraft during takeoff and landing is authorized by Task 11 of BuAer Contract NOa(s)

7866. Two 50-lb. thrust motors using liquid oxygen and alcohol have been built and static-tested in preparation for installment on a G & A Aircraft type AB-37B three-bladed rotor (37). A special whirling arm test stand was built for the preliminary tests, but a new one will be required to mount the actual test rotor blades. Final tests will be conducted as soon as this is available. This work is being continued under Task 8 of BuAer Contract Noa(s) 8540.

c. Task 12 of BuAer Contract NOa(s) 7866 was issued to cover the theoretical investigation of the problems which may be encountered in the combustion of liquid oxygen and gasoline or alcohol at controllable chamber pressures between 2,000 and 3,500 p.s.i. The flame temperature is required to stay within safe limits, and experimental investigations are required to verify the theoretical analysis. Other propellants, such as H_2O_2 and the boron compounds, were also to be considered, but to a lesser degree. This investigation is complete, and a report has been submitted.[7] A proposal for further development has been submitted to the Naval Aircraft Factory, Philadelphia, Pa.

d. Project "A" of BuAer Contract NOa(s) 8239 requires the design and erection of a large rotating test stand capable of holding a complete 6,000-lb. thrust liquid oxygen-alcohol rocket engine and tanks. This stand provides for locating accurately (with respect to the aircraft installation) the tanks and lines exactly as in the D-558 (Douglas transonic test airplane) and then rotating the stand about the pitching axis of the airplane as the unit is operating. This stand is also capable of handling thrusts up to 20,000 lbs. and is now in operation with the 6,000-lb. thrust unit mentioned above.

e. Project "B" of BuAer Contract NOa(s) 8239. This project calls for the development and testing of a prototype rocket engine (A6000C4) based on the motors and pumping system developed under BuAer Contract NOa(s) 7866 (36). This power plant consists of four 1,500-lb. liquid oxygen-alcohol-water rocket motors designed to be independently or simultaneously operated. The pumping system (designated 6M-325C) consists of four parts: a liquid oxygen pump, an alcohol-water pump, a turbine, and a combustion pot for driving the turbine. James Coolidge Carter of Pasadena, California, a pump consultant, designed and built the two pumps on a subcontract. The turbine wheel and support were taken from a standard aircraft turbo-supercharger, and Reaction Motors developed the combustion pot and all necessary controls for the operation of the complete system. This pumping system is required to operate for fifteen minutes without major repair or replacement of parts.

The prototype is near completion and will be installed on the special test stand described above for this unit for final testing. The rocket end of the power plant is very similar to the unit used in the XS-1 airplane and which was originally developed on Navy funds under contract NOa(s) 7866 (Fig. 26).

f. Project "C" BuAer Contract NOa(s) 8239. Upon completion of the acceptance test under Project "B" above this assembly (A6000-C4 motors and 6M-325C) pump assembly) shall be given an endurance test of five hours, making a complete and accurate report of all replacement parts used.

g. Project "D" BuAer Contract NOa(s) 8239. This phase requires the fabrication, assembly, and testing of seven complete A6000-C4 rocket engines and 6M-325C propellant pumps, and further to supply fourteen unassembled rocket engines and pumps with 300% spare pump bearings. Work on this project is under way and will approach completion at about the same time as the final acceptance tests.

h. Project "E" BuAer NOa(s) 8239. This phase requires the preparation of operation handbooks, overhaul manuals, and parts lists for the complete rocket unit described above. This work is under way and should be completed simultaneously with the prototype.

i. Task 1; BuAer Contract NOa(s) 8540. This task requires a detailed engineering study of turbopump units for propellant injection in rocket engines of the type described immediately above under paragraph *e*, and having ratings between 10,000 and 100,000 lbs. thrust in increments of 10,000 lbs. Experimental investigations shall be conducted to substantiate the theoretical calculations insofar as possible. The study shall include preliminary drawings and weight breakdowns. The study is under way.

j. Task 2, Phase 1, BuAer Contract NOa(s) 8540. A manually-operated variable thrust control is required for the A6000-C4 rocket engine.[8] The throttle shall be required in a test demonstration to vary the thrust on each individual cylinder from 1,200 to 1,800 lbs. by throttling the propellant feed pressure. The preliminary design of the equipment is complete and the parts are under construction. The basic principle is a variable restriction in the propellant flow lines.

Phase 2 requires continued development of the A6000-C4 rocket engine as found necessary from flight or static tests. This work will be carried on throughout the duration of the contract.

[7] "Gas Generators for Catapults," Reaction Motors, Inc., Report No. 186N-1, dated 10 October 1946. *Confidential.*

[8] "A Manually-Operated Variable Thrust Control for Liquid Rocket Engines," Reaction Motors, Inc., Report No. 183N. *Restricted.*

k. Task 3, BuAer Contract NOa(s) 8540. The further development of the 6M-325C liquid oxygen-alcohol turbo pump assembly with the gas generator and accessories which may be found necessary as a result of operation of this equipment with the A6000-C4 rocket engine. The objective shall be to increase the operational life of the turbo pump to one hour, decrease the overall length of the unit to 19 in. and reduce the propellant consumption to 1.10 lbs. per second. This work is proceeding along with the overall development of the D-558 power plant.

m. Task 5, BuAer Contract NOa(s) 8540. This work is a continuation of Task 10, BuAer Contract NOa(s) 7866. The original assignment called for an engineering study to provide the following:

1. Continuous variable thrust to meet any power requirement from minimum thrust to full rated thrust by remote control.
2. Automatic mixture ratio control of the two propellants in rocket engines.
3. Maintain a constant thrust setting by compensating for change in fuel flow rate, altitude, and air speed. Base this study on liquid oxygen-alcohol pump-fed system. Other bi-propellant systems should be considered.
4. Experimental investigation may be undertaken.

This study has been completed (34). The specifications under the present task for the construction of this control have been modified to read:

1. Constant thrust with varying altitude and air speed.
2. Constant air speed or Mach number by varying thrust from maximum rated thrust.
3. Constant mixture ratio within a range of approximately $\pm$ 4%.
4. The control device shall be designed for the A6000-C4 rocket engine but shall be applicable in principle to other rocket engine designs with minor modifications. Thrust shall be varied from 60% to maximum thrust. Specific impulse shall be at least 92% rated I_{sp} at 75% thrust and not less than 86% at 60% thrust.

The preliminary design of this equipment is complete and parts have been released for construction.

n. Phase 2, Task 6; BuAer Contract NOa(s) 8540. This phase covers the evaluation of anhydrous hydrazine (N_2H_4) and liquid oxygen as rocket propellants in a 50- to 200-lb. thrust rocket motors and at a chamber pressure of 300 p.s.i.a., utilizing water cooling. Specifically the evaluation shall cover the following:

1. Determine as a function of mixture ratio: effective exhaust velocity, characteristic velocity, coolant in and out temperatures and coolant flow rate.
2. Evaluate the following in regard to the fuel: regenerative cooling, ease of handling, stability, and storage problems.
3. Compare actual jet velocities with calculated values, and compare overall heat transfer for each mixture ratio with the same or a similar motor operating on oxygen-alcohol or acid-aniline.
4. Determine melting points for anhydrous hydrazine, hydrazine hydrate, and mixtures in the range of 100% hydrazine to 100% hydrazine hydrate.

Hydrazine for this evaluation was procured from the Fairmount Chemical Company of New York. Progress in this work is reported in bi-weekly progress reports to BuAer.

o. Contract NOa(s) 8531 with the Bureau of Aeronautics calls for a single cylinder 20,000-lb. thrust liquid oxygen-alcohol rocket engine. This engine is for the G. L. Martin high altitude sounding rocket being built for the Office of Naval Research. The prototype will be required to demonstrate a specific impulse of 215 for a period of 75 seconds. This system includes a turbopump propellant supply. The weight of the motor assembly is 290 lbs. and the pumping unit 125 lbs.

The preliminary design of the rocket is about 90% complete and the pumping system 50% complete. Some construction is under way. Ten complete units will be required by Martin.

p. Under a subcontract to Consolidated Vultee Aircraft Company, Reaction Motors is building an 8,000-lb. thrust swivel mounted liquid oxygen-alcohol rocket motor. The 8,000-lb. thrust will be broken down into four 2,000-lb. motors. A turbopump propellant injection system designated the 8M400-C will also be required, along with the gas generator and necessary controls. The system shall demonstrate a specific impulse of 215 seconds for a duration of 60 seconds. The motor assembly will weigh approximately 137 lbs. and the turbopump 114 lbs. Ten complete systems will be required upon approval of the prototype.

The prototype is 80% complete. Progress is reported monthly to CVAC and to the AAF.

q. General Electric Company purchase order 83465 with Reaction Motors. Requires the fabrication of the following equipment:

1. Two 750-lb. rocket motors for liquid oxygen and alcohol.
2. Two 50-lb. nozzles with helix and baffle.
3. Four 220-lb. nozzles with helix and baffle.

4. Four 350-lb. nozzles with helix and baffle.
5. Two 750-lb. nozzles with helix and baffle.
6. Two 1500-lb. nozzles with helix and baffle.

C. Hydrogen Peroxide Systems

1. AEROJET ENGINEERING CORPORATION.

a. Under the sponsorship of the Navy Bureau of Ordnance Contract NOrd-9768 Aerojet Engineering Corporation has conducted a review of the available literature on concentated Hydrogen Peroxide. Reference (7) is a report on this review and covers data on the physical properties, stability, storage and handling of this material. The report also covers the theoretical rocket motor performances of various hydrogen peroxide propellant systems.

b. Army Air Forces Contract W-33-038-ac-14835 has been let to determine the characteristics of the German ME-163 (Walter 109-509A) hydrogen peroxide-hydrazine hydrate rocket engine. The motor has been set up to be fired separately as a pressurized unit to insure its proper functioning. The pumps have been tested separately, pumping fuel, but the turbine was driven by steam instead of the combustion pot. The engine will have been set up and run as a complete unit by April 15, 1947. Mr. R. C. Stiff, who visited the Walter Works in Germany and conducted some demonstration tests there, has been acting as a consultant on this project.

2. JET PROPULSION LABORATORY, CALIFORNIA INSTITUTE OF TECHNOLOGY.

a. An active program is under way on a 200-lb. thrust scale to determine design and performance data for peroxide-ethanol and peroxide-nitromethane motors. Upon completion of this work, it is planned to initiate work on units using hydrogen peroxide with high performance fuels. These will possibly include liquid ammonia, hydrazine, methyl amine, and hydrogen.

Hydrogen peroxide has rather severe limitations as a coolant, and some work has been done on possible solutions of this problem. One of these is the use of refractory insulating liners. Another is the use of an annular chamber in which low temperature gases insulate the hot gases from the cooled wall.

3. NAVAL AIR MISSILES TEST CENTER, PT. MUGU, CALIFORNIA.

a. TED-PAU-PP-206, previously mentioned under the acid oxidizer section (p. 30), is the BuAer project order number set up at NAMTC for the familiarization and testing of captured German rocket equipment. Two such power plants, which require H_2O_2 for the oxidizer, are available. These units, listed below, are being prepared for test and will be run as soon as the test stands have been set up.

1. ME-163, airplane HWK109-509A Walter Works hydrogen peroxide rocket power plant. This unit is similar to the one being tested by Aerojet as mentioned above, but it is a slightly different model.

2. Hot ATO unit for bomber aircraft jet assisted takeoff. This unit was also developed by the Walter Works of Kiel, Germany. It develops 2,200 lbs. thrust for 42 seconds, using H_2O_2, gasoline and sodium permanganate as propellant. A specific impulse of 172 is realized.

4. REACTION MOTORS, INC.

a. Task 7, BuAer Contract NOa(s) 7866 calls for the study and testing of the German HS-293 missile rocket power plant (35). This power plant utilizes H_2O_2 as the propellant and potassium permanganate as the catalyst for decomposition. The Walter Works developed this unit and it is known as a cold type unit, since the principal exhaust product is steam. The motor develops about 1,300 lbs. of thrust for a period of ten seconds with a specific impulse of slightly over 100. The test project is complete.

b. Task 9, BuAer Contract NOa(s) 7866 required the preparation of a complete set of drawings of the turbopump unit of the German ME-163 hydrogen peroxide rocket power plant. This project is complete and the drawings are available in the Bureau of Aeronautics, Navy Department.

c. Task 10, BuAer Contract NOa(s) 8540. This task is for the conducting of a theoretical and experimental evaluation of a gas generator and liquid rocket using H_2O_2 of 90% concentration as a propellant. The work is divided into three phases as follows:

1. The testing of multi-propellant type rocket engines and gas generators of German design using 80% and 90% H_2O_2 with or without other propellants.

2. Limited development and testing of an experimental rocket engine of new design or modified German design using 90% concentrated H_2O_2 and alcohol as propellants.

3. Preliminary development and test of a gas generator designed by the contractor for the operation of turbo pumping units using 90% H_2O_2 with or without other propellants.

All phases of this task are in engineering status and some engineering detail is available in bi-monthly progress reports to BuAer. This work is being held up somewhat by the urgency and greater priority of other work. Item 3, however, is well under way.

D. Nitromethane Rocket Systems and Development

1. AEROJET ENGINEERING CORPORATION.

a. AAF Contract W-33-038-ac-11757 calls for the development of an experimental nitromethane turbo rocket which consists of a main direct thrust rocket power plant of five 1,500-lb. thrust motors or a total of 7,500 lbs. of thrust. The status of this program is explained below. A turbine pumping plant has been built and pumps tested, using a steam-driven Mark XIII torpedo turbine wheel. The combustion pot, consisting of a primary chamber for nitromethane decomposition and a secondary chamber for mixing of the diluent with hot gases, is a double motor running at a chamber pressure of 900 p.s.i. with an L* of approximately 200. A diluent of 70% water and 30% alcohol (alcohol for anti-freeze only) is introduced in the second chamber for reducing the exhaust gas temperature. The design has been established at 1.3 lbs./second propellant flow rate, where 0.9 lb./sec. is nitromethane in the first combustion chamber and the remainder is diluent in the second chamber. This combination has been run satisfactorily and will deliver the necessary turbine gas for the 7,500 lbs. of total thrust propellant pumping power. The pumps operate at 19,000 r.p.m. and will deliver 260 gallons per minute at 850 p.s.i. An uncooled combustion pot has also been built and run, but not in combination with the complete setup. Its operation alone was considered satisfactory and it is felt that it will also be satisfactory in combination with the remainder of the system.

The main motors have not been satisfactorily developed since the contract was predicated on the results of Navy development work. Two runs with modified uncooled chambers were made for preliminary injector data and resulted in injector burnouts. Present design trends are based on a chamber pressure of 900 p.s.i. at L* between 225 and 325. The oxygen injectors cause considerable trouble and the spark plugs will not stand up under more than one operation. It is felt by the test personnel that a satisfactory motor can be developed in a reasonable time.

Continuation of this work is waiting for approval from the Rocket Section at Wright Field for a time extension of the contract.

A report covering the development to date is in preparation for delivery to the Air Forces.

b. BuAer Contract NOa(s) 7968 has been let for the development of the XCNLT-600, a 600-lb. thrust nitromethane turbopumping rocket power plant for the guided missile "Lark." The work on this contract is considered complete by Aerojet and the unit has been delivered to the Navy. The 200-lb. thrust cruising motor was demonstrated for five minutes and the 400-lb. motor for slightly less; both were firing simultaneously. The contract called for repeated starting and stopping of the 400-lb. motor, but due to ignition difficulties only the shutoff was demonstrated. The pumping system was operated for a total of six minutes continuously (1).

c. BuAer Contract NOa(s) 8511 has been let for the continued development of nitromethane rocket systems. Specifically, the task assigned is to design, build, and test a 1,000-lb. (variable down to 300 lbs.) thrust nitromethane rocket motor to operate for five minutes, and further to design, build, and test a variable pressure turbopump for operation with the above motor. This contract has just been received, and preliminary design is complete. Parts from the XCNLT600 Lark unit will be utilized on this contract.

d. U. S. Navy Bureau of Ordnance Contract NOrd. 9768. This project is not strictly in the field of rocket development but from a practical point of view the work is directly applicable to both turbopumping systems and rocket systems, and hence it is worth including here. The problem as assigned is for the development of a 1,500-h.p. nitromethane gas generator and turbine as a torpedo power plant; and further, to investigate the feasibility of detonation-proof tanks for nitromethane, and to study and test chemical and other means of ignition.

A combustion pot has been built and fired only as a primary chamber. It develops 800 lbs. of thrust at a chamber pressure of 600 p.s.i. At this point in the development the requirements were changed to an 850 h.p. generator, and the chamber design is being reworked. The design study was complete on the 1,500 h.p. turbine but has not yet been revised for the 850 h.p. job.

Approximately 500 tanks have been shot at with armor piercing, tracer, and incendiary bullets. The correlation between wall thickness and degree of confinement has been established, but sufficient information to establish quantitative values has not as yet been accumulated.

A severance trap has been built which will stop detonation propagation through a tube and has been successfully tested under static low pressure conditions. Further tests at high pressures, static and dynamic, and at elevated temperatures remain to be made.

Several ignition schemes which have been developed are listed below:

1. Chemical ignition, using acid and aniline and acid-alcohol on a 50-lb. motor has been successfully accomplished. Half the normal flow rate of the operating propellant is used for the ignition propellant.

2. A mixture of hydrogen peroxide and hydrazine hydrate (83%) has been successful in 50-lb. and 200-lb. motors. In both this case and the one above, a chamber pressure actuated switch cuts on the main

flow when the igniter comes up to pressure. One-fourth of a second is the normal delay.

3. A solid fuel igniter utilizing paraplex and ammonium perchlorate cast on a 1/4 in. molybdenum rod has been successful in motors up to 200 lbs. thrust. The total igniter varies from 2 in. to 7 in. in length and is 2 in. in diameter. It is ignited by black powder and burns for 3 to 5 seconds.

BuOrd has recently stopped work at Aerojet on this project in favor of the lithium-water propellant hydro-turbojet reported on this page.

e. A summary of Aerojet nitromethane developments is included here as a review of the status.

1. Uncooled 350-lb. 25-second motor.
2. Uncooled 750-lb. 20-second motor.
3. Water-cooled 350-lb. 5-minute motor.
4. Uncooled refractory liner 350-lb. 120-second motor.
5. Diluent cooled 30 h.p. gas generator, and 20,000 r.p.m., 1,000 p.s.i., 25 g.p.m. pumping unit, for 6 minutes.
6. Regeneratively cooled 400-lb. 4-minute motor.
7. Regeneratively cooled 200-lb. 5-minute motor.
8. 800-lb. uncooled motor.
9. 300 h.p. gas generator 3.5 in. diameter, 12 in. long, for driving a 19,000 r.p.m., 850 p.s.i., 260 g.p.m. pump for a 7500-lb. thrust motor.
10. Uncooled motors of two-minute duration may be built, and gas generators with temperature from 800°F to 3500°F may be built.

2. Jet Propulsion Laboratory, California Institute of Technology.

a. The nitromethane investigations at JPL have been directed towards the following items:

1. Effects of geometry and shape of combustion chamber.
2. Injection.
3. Effects of catalysts on combustion process. Chromium salts have recently been tried.
4. Pressure transients during the starting period.

These studies have led to the design and construction of a 200-lb. nitromethane motor with a 60-second duration. A refractory liner is used in the chamber section. The aim is to develop a satisfactory fully regenerative unit. It is felt that more information will be needed on nitromethane as a coolant; heat transfer coefficients and the possibilities of boiling heat transfer will have to be studied to make this practical.

b. Considerable work has been carried on in the field of the detonation characteristic of nitromethane, including firing at tanks, line detonation traps, etc. No additional work in this field is planned for the present.

E. General Propellant Studies Including High Energy Fuels and Systems

1. Aerojet Engineering Corporation.

a. Office of Naval Research Contract N6ori-10, Task Order I, has been let for research and development on hydropulse, and hydro-turbo jet power plants. This work is not directly applicable to the present survey; but certain portions of the work are of interest.

Chemical preparation and study of the high energy fuels are to be carried out on a laboratory scale. A small-scale rocket motor has been run, using lithium and water as propellants. This was done primarily to determine characteristics as a simple rocket motor propellant and for comparison with other propellant combinations. Testing is complicated by the fact that the lithium must be held at 325°F to keep it molten. This is not serious in actual application, since the heat of the combustion chamber may be used. Present data indicate that the above propellants utilized in a torpedo hydropulse motor would be more than 4 times more efficient than an equivalent power standard torpedo propulsion mechanism at 50 knots and increasingly better as forward velocity increases.

Ethyl aluminum dihydride ($C_2H_5AlH_2+(C_2H_5)$ ALH) and diethyl aluminum hydride mixed with ethyl aluminum sesquihydride are being manufactured on a small scale for test purposes. When burned with water in a hydropulse, calculations indicate that these propellants should be 3 to 5 times better than the liquid alloy of sodium-potassium which was used during early hydropulse development.

A laboratory setup has been prepared for making aluminum boro-hydride, and tests will be made to measure heats of reactions with various oxidizers. Other hydrides will be made in an effort to determine which will be the best.

It is proposed to make numerous tests with very high temperature propellants in order to develop cooling methods and metallurgy as well as determine the propellant characteristics. Controls and accessories will be developed as required by the peculiarities of each propellant system.

Progress reports on the status of this work are forwarded to the Office of Naval Research.

b. Under a subcontract with Consolidated Vultee Aircraft Corporation, Aerojet has conducted a study on rocket motors using the high energy fuels. Specifically the problem as undertaken was to calculate the performance of the boro-hydrides and hydrogen and oxygen propellant systems for rockets. The specific impulse was calculated and estimates of sizes and weights of rocket motors were included along with an outline of a program to develop the propellants considered.

This work is complete and a report was submitted to CVAC for inclusion in their report to the AAF Guided Missiles Laboratory at Wright Field under the AAF prime contract W-33-038-ac-14168.

2. General Electric.

a. See page 35 for G.E. work on liquid oxygen-diborane motors.

3. Jet Propulsion Laboratory, California Institute of Technology.

a. A program is being formulated on the study of very high exhaust velocities such as will be encountered with rockets utilizing nuclear energy. It is considered essential to secure data in this range to guide future development work. It can be shown that exhaust velocities of the order of 25,000 ft./sec. are obtainable using hydrogen as the working fluid raised to a temperature of 5,000°F

At the JPL it is contemplated to attain the high temperature by the electric heating of porous molybdenum with the use of hydrogen gas as the working fluid. Other fluids may be used later. Detailed studies will be made of the heat transfer problem. Attempts will be made to check the theoretical calculations for exhaust velocity. Engineering problems such as nozzle erosion will be studied. It is also planned to make spectroscopic observations in order to gain an insight into some of the internal processes.

4. National Advisory Committee for Aeronautics, Aircraft Engine Laboratory.

a. A theoretical study is being made to determine the performance of various oxidizers and fuels in different combinations. Details of this study are presented in Volume I, Part 2 of this Survey, entitled *Fuels*.

5. North American Aviation, Inc.

a. Under AAF Contract W-33-038-ac-14191, previously mentioned, the NAA Aerophysics Laboratory research section has been conducting a rocket propellant study. The work has been completely analytical to date. A 50-lb. motor based on a GALCIT design is planned for small-scale tests. Further information on this work is presented in Volume I, Part 2, entitled *Fuels*, of this Survey.

6. Naval Ordnance Test Station, Inyokern, California.

a. Projects No. 267 and 268 have been assigned for the development of a liquid rocket artillery missile having a velocity of 1,000 to 3,000 ft./sec. A determination will be made of optimum size and of conditions under which a change from solid fuel to liquid fuel becomes desirable. No restrictions have been placed on the choice of propellants except that it must be capable of being stored for at least two years in a sealed container, without decomposition or attack on the container. The container will be a part of the missile and must be expendable. For this reason, attempts will be made to use ordinary steel, and such metals as stainless steel, nickel, and chromium will be avoided.

b. An investigation of nitrogen peroxide and aniline as rocket propellants is under way. A small 350-lb. thrust motor ($L^* = 200$) has been built for testing this propellant combination. Nitrogen peroxide was selected because it is non-explosive, entirely stable on storage, and can be stored as a liquid under moderate pressure in ordinary steel cylinders. Under tropical storage conditions, the pressure in the tanks should not exceed 400 p.s.i.

7. Reaction Motors, Inc.

a. Phase 1, Task 6 of BuAer Contract NOa(s) 8540 has been established for the investigation and development of a rocket engine to use the high energy propellants. Primarily this involves diborane and other boron compounds. Tests will be conducted to determine the performance characteristics of these compounds as rocket propellants. Particular emphasis will be placed upon determining maximum specific impulse which may be expected, as well as solving the mechanical, thermal, storage, and handling problems which may develop during the course of the investigations. The contract requires the development of a rocket motor of from 100 to 400 lbs. thrust for one minute duration and a specific impulse of 300 seconds or greater.

This task also includes a design study and a preliminary design for two rocket engines, complete with turbopump feed systems, utilizing the most promising propellant combination as determined by the above investigation. The sizes of these two systems are 10,000 and 100,000 lbs. thrust.

F. Rocket Motor Cooling, Combustion Chamber, and Injector Studies.

1. Aerojet Engineering Corporation.

a. The problems of sweat or transpiration cooling of high temperature hydrogen oxygen motors have been discussed on page 31 of this report.

2. Jet Propulsion Laboratory, California Institute of Technology.

a. *Sweat Cooling.* A very extensive program is under way on basic studies of sweat cooling. Thus far dies have not been fabricated for actual applications to rocket motors. Preliminary experiments have been made successfully on the development of techniques for the hydrostatic pressing of such items as nozzles. Techniques have also been developed for measuring permeability and flow rates through various porous specimens using both gaseous and liquid test fluids.

b. Film Cooling. Although film cooling may be less efficient than sweat cooling, it is much less complicated and as a result the state of the art is more highly developed. Systematic tests have been carried out on a 1,000-lb. thrust scale to obtain design information on this type of cooling. Further, it is planned to investigate the basic aspects of film cooling by simplifying the problem to that of fluid ejection through a single hole, determining the spread of the film, etc.

One of the key theoretical problems as yet unsolved is that of heat transfer from a hot flowing gas to a metal wall through a liquid layer. This, together with the heat transfer problem in sweat cooling, where there is continuous mass addition to the boundary layer, are vital problems which should be attacked vigorously. Work on the latter problem is now under way.

c. Conventional or Regenerative Liquid Cooling. Application of present theories of convective heat transfer to the problems involved in a rocket motor is not very satisfactory for several reasons:

1. Coolant properties in the temperature range encountered are not very well known.
2. Direct application of heat transfer coefficient derived for cases of small heat flux to cases of high heat flux is questionable.
3. Boiling heat transfer is not understood.

To attack some of the unknown elements involved in the heat transfer problem, the JPL has initiated studies using a flame tube to obtain convective heat transfer data such as the inter relationship between Reynolds's, Prandtl's, and Nusselt's numbers and the viscosity ratio. Other similar tests are planned for a later date, using an electrically-heated tube. When the tests are completed on the straight tube, it is planned to extend the study to determine the effect of these factors in a curved tube, which is necessary for extrapolation of the data to actual regeneratively-cooled motors.

Further extension of this work will include boiling heat transfer problems such as the limiting value of the heat flux at which the conductance of a boiling film starts to break down, and its dependence on temperature, pressure, flow rate, etc.

d. Combustion Chamber Characteristics. A general understanding of features such as temperature distribution, composition distribution, velocity distribution, etc., is essential before it will be possible to establish logical methods for guiding combustion chamber design. In line with these ideas, a series of studies to determine these distributions in existing motors has been started. First experiments are being made in hydrogen peroxide motors with a transparent wall, using a calcium permanganate solution as the catalyst. The reason for this is that the temperatures are low enough to be easily measured. Later the work will be continued to include a peroxide alcohol motor, possibly an acid type motor, and a nitromethane type. It is hoped to be able to correlate these distribution measurements with systematic changes in methods of injection, combustion chamber shapes, etc. It is also planned to include in this program studies of erosion conditions inside the chamber.

e. Combustion Chamber Shapes. Studies have been continuing on tubular motors to include determinations of effects of length, pressure, and mode of injection on exhaust velocity, heat transfer distributions, etc. The chief penalty in the use of such motors appears to be the additional pressure drop. It has been found that to obtain performance equivalent to that at 300 p.s.i., a pressure of 325 p.s.i. is required at the chamber head in the tubular type motor. It has been found also that the combustion volume can be much smaller than initially believed.

Because of the pressure drop penalty, work is being initiated on "near" tubular motors, which differ from the complete tubular motor in that there is a "necking down" at a section between the tubular portion and the diffuser; in other words, a throat is formed at which sonic velocity must occur.

f. Injector Design. Work is in progress involving experimental studies of injectors to determine characteristics such as discharge coefficients, cavitation properties, geometrical spray patterns, and so forth. Droplet size distribution and velocity distribution are also being investigated.

g. Heat transfer, injector design, and L^* investigations are also under way for the liquid oxygen-methanol and ethanol systems as reported on page 34 of this report.

3. NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS, AIRCRAFT ENGINE LABORATORY, CLEVELAND.

a. The general problem of rocket motor cooling is being studied with particular regard at present to nozzle cooling.

1. An Aerojet 25ALD 1000 rocket unit (acid-aniline) has been rigged with water injection just upstream of the nozzle. The water is injected both tan-

gentially and coaxially with the chamber wall. Tests are being run with varying amounts of water, and later tests will be run injecting the same amount of water in the center of the chamber to determine its effect on performance. To date the performance is the same when based on propellant flow, but is reduced by about 5% (water injected weight) when the weight of the water is taken into account. Numerous thermocouples are located on the nozzle and chamber outer walls for temperature measurement.

2. Fundamental studies are also planned on the mechanism of heat transfer, and this work will be done on special test setups as well as in rockets. Gas and liquid film cooling will be studied, and also sweat cooling through porous wall chambers. It is expected that with the temperature reached with the high energy propellants, regenerative cooling will not be adequate and steps must be taken to guarantee the cooling problem solution before these propellants are tested.

4. NORTH AMERICAN AVIATION, INC.

a. The sweat cooling problem is being investigated from an analytical viewpoint. The available literature is being studied and experimental work may be undertaken upon the completion of the studies.

5. REACTION MOTORS, INC.

a. Task 4, BuAer Contract NOa(s) 8540. This investigation is for the purpose of developing suitable means of cooling a rocket engine when operating at temperatures in excess of those occurring in present motors. The development of material or cooling methods satisfactory for use with the high energy propellants will be undertaken. This includes a survey of the synthetic high temperature materials and porous metallic materials as well as other methods of cooling which may become apparent. Experimental investigations will be carried out in specially designed experimental rocket engines.

The efficiency of heat of flux at various liquid coolant pressures, densities, and flow rates at elevated combustion chamber temperatures is being investigated. It is felt that regenerative cooling cannot be made to work with the high energy propellants.

Progress reports are submitted bi-weekly to BuAer.

b. Task 9, BuAer Contract NOa(s) 8540. Phase 1 is a series of parameter studies on a 1,500-lb. thrust rocket motor to study the effects of mixture ratio, water dilution, fuel additives, and operating pressure on efficiency.

Phase 4 is a special research rocket motor for high pressure tests with provisions for temperature measurement in the motor jacket and pressure distribution measurement in the motor jacket and nozzle.

This work is under way and is reported in bi-weekly progress reports to BuAer.

G. Component Development and Miscellaneous Problems of Liquid Rocket Systems.

1. AEROJET ENGINEERING CORPORATION.

a. Bureau of Aeronautics Contract NOa(s) 8516. This contract is for preliminary design of the rocket test station, to be constructed at NAMTC, Pt. Mugu.

2. JAMES COOLIDGE CARTER, PUMP CONSULTANT, PASADENA, CALIFORNIA.

a. Mr. Carter has designed the liquid oxygen and alcohol-water pumps for the Reaction Motors 6,000-lb. thrust rocket engine. These pumps deliver approximately 14 lbs./sec. propellant flow each, at a discharge pressure of 325 p.s.i. An inlet pressure of 35 p.s.i. is used on the oxygen and 10 p.s.i. on the alcohol-water.

b. Mr. Carter has recently completed the design and construction of a set of white acid-furfuryl-alcohol pumps for a 6,000-lb. thrust rocket power plant for use on the Naval Air Missiles Test Center high altitude sounding rocket (Fig. 15).

3. JET SPECIALTIES COMPANY, DIVISION OF G. I. FARMAN INDUSTRIES, PASADENA, CALIFORNIA.

a. This company has been organized to develop and supply items peculiar to the jet propulsion industry. Among the items handled are valves, metal and plastic tubing, filters, hydraulic presses, plastic and synthetic materials for gaskets and "O" rings, and hydraulic test stands.

A special project for the development of Polythene and Polyethylene "O" rings has been under way for some time. These will be particularly useful in acid handling equipment.

4. NAVAL AIR MISSILES TEST CENTER, PT. MUGU, CALIFORNIA.

a. TED-PAU-PA-203. The noise and vibration studies of P/A power plants are virtually complete. Tests were run on the "Lark," "Gorgon," and "Loon" installations to determine the frequencies, if any, that might disturb the electronic guidance and control equipment. The report on this work will be available through the Radio and Electrical Section of BuAer.

5. NAVAL GUN FACTORY, WASHINGTON, D. C.

a. Aviation Ordnance Project No. 408 was assigned for the "Investigation of Rocket Blast Characteristics." This work is not strictly applicable to this report since it does not concern itself with development of rockets. It is felt however that the work is of such a character that the instrumentation developed may well be of interest to rocket development agencies.

The project purpose is to carry on applied research, both theoretical and experimental, into the exhaust blast characteristics of liquid fuel and double-base solid propellant rocket motors. It is planned to assemble, verify, and compile data for use in the design of aviation ordnance rocket conveying, supporting, launching, and control equipment. No attempt to improve on existing motors or fuels is anticipated.

The immediate objectives are in connection with 3½ in. and 5 in. fin-stabilized and 5 in. spin-stabilized solid fuel rockets. Instruments are being developed to measure:

1. Maximum pressure and pressure gradients in free air surrounding the exhaust.
2. Maximum temperature and temperature gradients in free air surrounding the exhaust.
3. The temperature and temperature gradients in the exhaust flame itself.
4. Variations with time of temperature and pressure in the surrounding air.
5. The temperature and pressure developed on the surface of local structure in the field of blast.
6. Sound frequency spectrum analysis and energy output distribution in the region up to 21 kilocycles.
7. The mechanical condition of each rocket motor before firing and the alignment of the jets.
8. The dynamic condition during firing including a measurement of axial thrust, torque, and assymmetric loading about the center of gravity.

The long-range objectives include:

1. Maximum pressures and pressure gradient in tubes and ducts both straight and with bends.
2. Maximum temperature in and on tubes and ducts.
3. The effects both on the rocket and on the duct, of muffled type exhaust. This work will include temperature, pressure, and duct thrust measurements.
4. The velocity of the gases in ducts and in free air.
5. The radiated energy distribution in the frequency range from 21 kilocycles through several megacycles.
6. Investigation of shock wave phenomena. This work will include the development of possible counter shock means.
7. Determination of the results on turbulence and temperature of using rough wall exhaust tubes of flexible type.
8. Gas sampling techniques and equipment for measuring gas products of combustion. This apparatus will be designed for use in confined wing spaces as well as open air.
9. Pressure and blast effects surrounding muzzles and breeches of both normal and recoilless guns.

The work has been divided into 42 projects, of which only 21 are being pursued due to lack of personnel. At this point only a portion of the instrumentation is complete and first tests await completion of the remainder.

6. North American Aviation, Inc.

a. The general problem of jet vane steering devices is being considered. A literature survey (German material) was made, and some aerodynamic tests have been run in the small supersonic wind tunnel. Life tests on vane materials have been made in the exhaust of an Aerojet 3SALDW-1500 JATO unit.

7. Reaction Motors, Inc.

a. Task 3, BuAer Contract NOa(s) 7866. The original assignment of this task required the development and prototyping of a liquid nitrogen evaporator for pressurization of propellant tanks adequate to operate the 6,000-lb. thrust liquid oxygen-alcohol water unit developed under Task 1 of this contract.

The equipment was built by Linde Air Products according to specifications set up by R.M.I. The unit operated two 1,500-lb. motors satisfactorily, but it was necessary to modify it to operate four. The complete unit was extremely heavy and was designed with a large safety factor, hence it is good for ground test stand operation only. The unit could conceivably be redesigned, however, for aircraft use, but there is small probability of saving in overall weight over a pressurized system. The density of the system, however, would be increased materially, and this is important from drag considerations.

This work is being continued under Phase 3 of Task 9 BuAer Contract NOa(s) 8540.

b. Phase 2, Task 9, BuAer Contract NOa(s) 8540. The investigation of both nitrogen and helium gas-feed systems to determine the proper capacity of high pressure tanks taking into account all associated problems.

c. Phase 5, Task 9, BuAer Contract NOa(s) 8540. The systematic studies of regulators and valve characteristics for determination of flow capacity, pressure drop, and regulation.

d. Portable acid aniline servicing equipment is discussed on page 31 of this report.

e. A large rotary rocket test stand is discussed on page 36 of this report.

8. White Sands Proving Ground.

a. The work at White Sands, Ft. Bliss, Texas, is under the joint direction of U. S. Army Ordnance and the General Electric Company in connection with Project HERMES. No development work is under way on rockets except incidental work in improving or changing the A-4 power plant which becomes necessary as part of the regular launching program. The development of the HERMES II missile, in which the A-4 acts as a booster, may occasion some further development.

Table V. Summary of Current American Rocket Engine Development.

No.	Thrust	Manufacturer	Designation	Duration in secs.	Propellants	Specific Impulse	Fuel Feed System	Chamber Pressure	Status	Use	Sponsoring Agency
1	50	RMI	—	Cont.	LIQ.O_2-GAS.	—	Press.	—	Test	Rotory Wing Drive	Navy
2	400	RMI	—	60	Diborane+Oxid.	300+	Press.	—	Test	Experimental	Navy
3	620 (220+400)	Bendix	CML-4N	180	MA-MEA	—	Pump	—	Test	LARK Missile	Navy
4	620 (220+400)	NAMTC	CML-4N	180	MA-MEA	—	Pump	—	Test	LARK Missile	Navy
5	1,000	Aerojet	—	—	RFNA-AN	—	Press.	—	Engineering	HASR	Boeing
6	1,000	Aerojet	—	—	LIQ.H_2-LIQ.O_2	—	Press.	—	Engineering	Experimental	Navy
7	1,000/300	Aerojet	—	300	NM	—	Turbopump	—	Engineering	Experimental	Navy
8	1,500	Bell Aircraft	—	—	LIQ.O_2-ALC.	—	Press.	—	Engineering	METEOR Test Missile	Navy
9	2,000 Var.	RMI	—	—	MA-MEA	—	Press.	—	Construction	Experimental	Navy
10	2,600	Aerojet	AJE-1-A	45	RFNA-AN-ALC	193	Press.	300	Engineering	XASR-1, NATIV Miss.	Navy
11	4,000	Curtiss Prop.	—	60	LIQ.O_2-Gas.-H_2O-Alc.	—	Turbopump	—	Test	XB-45 JATO	Army
12	4,000	Aerojet	60ALD4000	60	RFNA-AN-ALC	194	Press.	—	Test	B-45 JATO	Army
13	4,000	Aerojet	60AL4000	60	RFNA-AN-ALC	194	Pump	—	Design Study	XB-47 JATO	Army
14	6,000	NAMTC	—	93	NA-ALC	217	Pump	600	Design Study	NAMTC-HASR	Navy
15	6,000	RMI	A6000C4	Cont.	LIQ.O_2-ALC.H_2O	194	Turbopump	215	Test	D558 Research Airp.	Navy
16	6,000	M. W. Kellogg	—	—	NA-N_2H_4	—	—	—	Design Study	Experimental	Army
17	7,500	Aerojet	—	—	NM	—	Turbopump	900	Test	Experimental	Army
18	8,000	RMI	8000C4	60	LIQ.O_2-ALC-H_2O	215	Turbopump	—	Construction	CVAC Missile	Army CVAC
19	10,000	RMI	—	—	Boron Compds.	—	Turbopump	—	Design Study	Design Study	Navy
20	10,000	Curtiss Prop.	—	120	LIQ.O_2-JP1-Alc.-H_2O	—	Turbopump	—	Engineering	Experimental	Army
21	14,000 (4,000+10,000)	M. W. Kellogg	—	—	NA+Hydrocarbon	—	Press.	—	Design Study	Experimental	Army
22	15,000 (10,000+5,000)	Curtiss Prop.	—	172	LIQ.O_2-JP1-Alc.-H_2O	185	Turbopump	—	Engineering	XS-2 Aircraft	Army
23	16,000	Curtiss Prop.	—	—	LIQ.O_2-JP1-Alc.-H_2O	—	Turbopump	—	Design Study	XB-47 JATO	Army
24	16,000+1,200	Curtiss Prop.	—	150-300	LIQ.O_2-JP1-Alc.-H_2O	—	Turbopump	—	Design Study	XP-91 Aircraft	Army
25	16,000	General Electric	A-1	—	LIQ.O_2-Alc.	—	Press.	—	Design Study	HERMES A-1 Missile	Army
26	20,000	RMI	—	75	LIQ.O_2-Alc.	215	Turbopump	—	Engineering	Martin HASR	Navy
27	20,000	GALCIT	Corporal F	60	RFNA-AN	—	Turbopump	300	Test	Corporal F Missile	Army
28	40,000	Aerojet	90AL40,000	90	NA-Alc.	—	Turbopump	—	Engineering	Experimental	Army
29	60,000	Curtiss Prop.	—	—	LIQ.O_2+Hydrocarbon	—	Turbopump	—	Design Study	Experimental	Army
30	70,000	M. W. Kellogg	—	82	LIQ.O_2-N_2H_4	270	Turbopump	300	Design Study	Design Study	Army
31	100,000	RMI	—	—	Boron Compds.	—	Turbopump	—	Design Study	Design Study	Navy
32	100,000	Curtiss Prop.	—	—	LIQ.O_2+Hydrocarbon	—	Turbopump	—	Design Study	Experimental	Army
33	100,000	Aerojet	—	—	LIQ.O_2-LIQ.H_2	—	Turbopump	—	Design Study	Design Study	Navy
34	290,000	M. W. Kellogg	—	82	LIQ.O_2-N_2H_4	270	Turbopump	300	Design Study	Design Study	Army
35	300,000	Aerojet	—	—	LIQ.O_2-LIQ.H_2	—	Turbopump	—	Design Study	Design Study	Navy

VI. FACILITIES AND PERSONNEL

A. Aerojet Engineering Corporation, Azusa, California

1. Personnel. A total of over 500 people is employed. The research division has 85 people, including 40 mechanics, 4 stenographers, 40 engineers and scientists. The engineering group consists of approximately 94 engineers and draftsmen. The remainder of the staff is comprised of administrative, clerical, mechanical, and production personnel.

2. Facilities.

a. The Aerojet facilities, other than test stands, consist of a machine shop, supply and shipping warehouses, administrative offices, engineering department and drafting room, production buildings for solid propellant units, chemistry laboratory, hydraulic laboratory, research department offices and separate assembly and machine shop and numerous other small buildings which are used by the above groups for additional space. A considerable portion of all of the facilities is devoted to solid propellant rocket work.

b. A total of fourteen test stands, one ring channel, and one detonation pit are available for liquid and solid propellant rocket work. Two test stands are used exclusively for solid propellants. The largest solid propellant motor tested to date is the 2AS66000. Six stands are assigned to the research division and six to the engineering department for liquid propellant tests. Essentially all of these stands are of what may be called conventional design. The rocket motors or systems are mounted on suitable structural steel frameworks with provisions for thrust measurement. Controls and instrumentation are located in blockhouses which provide adequate protection from and view of the apparatus under test. It will be possible to test liquid propellant motors up to 20,000 lbs thrust, with present instrumentation and controls, as soon as a test stand and tanks planned for immediate construction are completed. The ring channel is used for hydropulse testing and the detonation pits for propellant shock and sensitivity tests.

The company and test stands are located approximately one-half mile from the nearest residential area, and there has been no serious objection to the noise of rocket testing.

B. Bell Aircraft Corporation, Buffalo, New York

1. Personnel. Sixteen engineers or scientists and fifteen technicians, mechanics, and shopworkers are being utilized to carry out the Bell rocket program for Project METEOR.

2. Facilities. Present facilities consist of: (1) four test stands for liquid propellant motors of 10,000 lbs., 6,000 lbs., 1,500 lbs., and 250 lbs. thrust; (2) one turbine pump test stand, and (3) one small test stand for turbine combustion chamber and control studies. All machine shop work is done in the experimental aircraft shop, and hydraulics testing is done in conjunction with the rocket test stands. A steam ejector type supersonic wind tunnel is contemplated as being necessary for the successful completion of the Bell portion of Project METEOR.

The Bell plant is located in the country at an airport and is thus ideally situated from the standpoint of testing noises and flight test facilities.

C. Eclipse-Pioneer Division, Bendix Aviation Corporation, Teterboro, New Jersey

1. Personnel. There are approximately twenty people of engineering or scientific caliber in the research engineering division. Only a small portion of these is directly engaged in the research and design of the rocket propellant pumps described elsewhere in this report, but the entire group is available for consultation and advice.

2. Facilities. The Research Engineering Test Laboratory is located in a small two-story building of approximately 75′ x 75′. This building is equipped with a small shop in which testing is conducted on turbines and pumps, air supply from four naval torpedo flasks is used to provide an air jet of about Mach number 2.0 for thirty seconds in a 2-inch line. This drives the turbines for cold tests. This building also contains an altitude chamber for simulating altitudes up to 50,000 ft. The shops of the main plant are available for fabrication of the test articles.

All hot tests of the pumps and turbines will be conducted at the test facilities of Reaction Motors, Inc., at Lake Denmark, New Jersey.

D. Consolidated Vultee Aircraft Corporation, Vultee Field, Downey, California

1. Personnel. There are approximately fifteen engineers and draftsmen assigned to the ducted rocket program and about five technicians or mechanics.

2. Facilities. Two blowers are available for work; 3,000 c.f.m. of standard air is delivered at 100 p.s.i. gauge by a two-stage reciprocating compressor. The other compressor (Sutorbuilt-Roots) will deliver 15,000 c.f.m. of standard air at 10 p.s.i. gauge. The latter is being used for the ducted rocket tests.

The blowers are located in a large open space of many acres, and are about 500 ft. from the main factory building. They are sufficiently distant from settled communities so that there will probably be no serious complaint about the noise.

E. Curtiss-Wright Corporation, Propeller Division, Caldwell, New Jersey

1. PERSONNEL. Fifty-eight people in all are now directly engaged in liquid propellant rocket development at Curtiss. This number may be broken down into two administrative engineers, thirty-five engineers, ten mechanics, and machinists, five test engineers, and six secretarial and clerical. It is anticipated that this group will grow to between one and two hundred people, depending upon facilities available.

2. FACILITIES.

a. Present. The engineering and drafting rooms occupy approximately 4,300 sq. ft. with considerably more space available, if necessary. The machine, mock-up, and electronic-hydraulic shops occupy about 5,900 sq. ft. in a former AAF training school barracks separate from the main plant.

The test facility is housed in two small buildings located on the Caldwell Wright Airport adjacent to the Propeller Division flight test hangar, about one mile from the main plant. The Control and Observation building of approximately 700 sq. ft. is of wooden construction and houses a small shop as well as a single control room for the two test stands. An open court separates this structure from a fireproof Rocket Test building of 1,300 sq. ft. which houses two test stands and miscellaneous components test rigs.

The smaller test stand is complete with pressurized tanks for testing motors up to 1,500 lbs. thrust and the larger stand is built to take motors up to 10,000 lbs. thrust. The latter stand is interesting from the point of view of propellant supply. The fuel is supplied by a motor-driven horizontal type centrifugal pump which will deliver the necessary flow-rates at pressures up to 700 p.s.i. A motor-driven vertical pump pressurizes the coolant supply. The liquid oxygen may be supplied from the storage tank by a centrifugal pump as soon as it is built and tested. In the interim the oxygen supply is pressurized by helium gas. In addition, it is planned to incorporate a specific impulse meter into this stand. The instrument will be set up so as to divide the thrust by the total flow rate with an electrical circuit. Thus it will be possible to determine rapidly the variation in specific impulse with mixture ratio by manually varying the propellant flow rates. Some objection to the noise from the local populace is anticipated and sound suppressing equipment is to be installed.

b. Proposed. A contract is under negotiation with the AAF for a test facility to be installed at Picatinny Arsenal, Dover, New Jersey. Present plans call for the following equipment at this location.

1. Two 20,000-lb. test stands.
2. One 30,000-lb. test stand (angle firing).
3. One 60,000-lb. test stand (angle firing).
4. One component test facility (pumps, valves, etc.) for systems up to 60,000 lbs.
5. Duplication of Items 3 and 4 and joint use of Item 2 for other contractors is being considered.
6. Separate pump sheds will be set up from which the propellants will be fed by steam-driven turbine pumps to the various test stands.
7. Block House for control of Items 1, 2, and 3.

It will require approximately one and one-half years to construct this facility, if the contract is granted.

F. General Electric Company, Schenectady, N. Y.

1. PERSONNEL. Approximately one hundred people are employed at the Malta test station, where all rocket motor designs and tests are undertaken. An additional average of twenty people is employed at the main plant in Schenectady, where all fabricating and special problems are handled.

2. FACILITIES. The Malta Test Station of the General Electric Company "Project Hermes" is located 22 miles north of Schenectady and 1½ miles inside of the border of a very large forest reserve. It is ideally suited from the noise and disturbance standpoint, but 1½ hours are lost daily per man traveling to and from work.

Three test stands are presently in operation. Test stand No. 1 will handle motors up to 1,800 or 2,000 lbs. thrust with present propellant tanks but can be readily revised to handle larger motors. Test stand No. 1a is an appendage on test stand No. 1 and uses the same instrumentation. It is designed primarily for small motors using the high energy propellants.

Test stand No. 2 is used for testing motors up to 4,000 lbs. thrust at present, and new propellant tanks will be installed to handle motors up to 8,000 or 10,000 lbs. The motors in the above three stands are fired horizontally. Flow rates are measured by a running propellant tank weight reading which is later corrected for the pressurizing gas weight. All standard measurements are taken and continuously recorded through the runs.

Two additional vertical firing test stands are under construction. These stands will handle 50,000-lb. motors for 50 seconds or larger motors for less time. An additional test area is being planned for flow studies in determining accurately gas requirements for pressurizing propellant tanks.

The main building at the test site is 95% complete and will contain the following functions: offices, drafting room, chemistry laboratory, instrument shop (for repair and calibration, and a small amount of development), photographic laboratory, hydraulic laboratory, and assembly shop.

A small machine shop is available for doing maintenance and test stand work, and two portable Army machine shops on trucks are used for on-the-spot work. The water supply building is being used at present as a flow study and hydraulic laboratory, but this will soon be transferred to the main building.

Two 11-ft. diameter spheres are to be installed for liquid oxygen and liquid nitrogen storage. The oxygen and nitrogen will be evaporated and compressed for propellant tank pressurization. Liquid oxygen is transported from the sphere to the test pits in 100-gallon Linde buggies. Alcohol fuels are being stored in standard Army gasoline trucks.

The facilities of the main plant in Schenectady are available for all fabrication work in addition to the various laboratories typical of such large organizations.

G. Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California

1. Personnel. Five hundred people in all are employed at the Laboratory, of which one hundred are of scientific or engineering caliber, and the remainder are technicians or mechanics.

2. Facilities. Eight small test cells, each containing two 1,000-lb. thrust test stands, are available for liquid propellant rocket testing. In addition, a stand for motors of 20,000 lbs. or over is located at the Army Air Base at Muroc, California.

Besides the test stands, offices, and engineering department, the Jet Propulsion Laboratory is well equipped with supplementary facilities. These include a machine shop, a complete hydraulic laboratory, an instrument shop, a chemical laboratory, and an excellent metallurgical laboratory. The latter contains an electron microscope, an X-ray diffraction machine, two electric furnaces which will reach 4,000°F., and a large hydraulic compressor for forming or molding porous-walled combustion chambers and nozzles of powdered metals. A fluid mechanics laboratory for the investigation of fundamental problems, and an aerophysics laboratory for missile guidance and control problems are also available.

In addition, funds are available for the construction of a large supersonic wind tunnel which will reach Mach number 4.6 in a 17 in. by 20 in. working section. It is estimated that this facility will be in operation within the next eighteen months.

H. M. W. Kellogg Company, Jersey City, N. J.

1. Personnel. About twenty engineers and scientists and five technicians have been assigned to the Rocket Engineering Division. An indeterminate number of mechanics and machinists are employed in fabricating and assembling test equipment. The personnel of the Special Projects Department of the M. W. Kellogg Company, about one hundred fifty people, are more or less directly concerned with rocket motor development.

2. Facilities. Two test stands are available for rocket testing. One stand is designed for motors up to 200 lbs. thrust and the other for motors in the 10,000-lb. thrust range. Standard types of Bourdon gauges and thermocouples are used as part of the instrumentation in both test stands. A machine shop is available along with facilities for assembly work, hydraulic testing, and pump and turbine testing.

The main activity of the M. W. Kellogg Company is the engineering of chemical process plants, and hence facilities are available for construction of heavy chemical equipment. Well-equipped general chemical and metallurgical laboratories are used by the rocket division.

The plant is located on the outskirts of Jersey City, adjacent to Newark Bay on a large plot of ground in an industrial district. Noise and handling of dangerous materials should offer no serious difficulty.

I. Aircraft Engine Laboratory, National Advisory Committee for Aeronautics, Cleveland, Ohio

1. Personnel. Twelve scientists, including chemists, physicists, electrical, chemical, and mechanical engineers, etc., are actively engaged in the rocket program. Six mechanics are available at the test site for testing operations.

2. Facilities. The general facilities at NACA, Cleveland, are very extensive and are tailored to power plant work. There are some fifty buildings located on a large tract of land adjacent to the Cleveland Airport. They have their own hangars and can flight-test airplane-rocket combinations such as JATO, rocket planes etc. Main shop facilities are elaborate and they are capable of prototype construction of large units.

The laboratory has four liquid rocket test stands. A central instrument room is used to house the more expensive and elaborate recording instruments, and this room is used for all stands. Pressure measurements are recorded in the test cells. Two propellant storage buildings are located nearby, as well as a small assembly shop and remote control room for dangerous tests. The rocket test site is located at some distance from the main buildings and earth mounds are used to provide protection between buildings and to act as sound deflectors. The site is sufficiently isolated, however, so that noise should not become a serious problem.

J. U. S. Naval Gun Factory, Washington, D. C.

1. PERSONNEL. The exact number of personnel assigned to Project No. 408 "Investigation of Rocket Blast Characteristics" is variable depending on personnel available and status of work. The problem is primarily one of instrumentation, and so these people are not necessarily of direct interest to this report.

2. FACILITIES.

a. Mobile Laboratory. Because of Gun Factory regulations prohibiting the storing or firing of large quantities of potentially explosive materials it is necessary that most measurements be made at field stations. In order to facilitate the assembly and operation of the complex equipment essential to blast and pressure wave research, a van with trailer has been procured and is being prepared as a mobile blast research laboratory. This trailer will contain all of the instrumentation required, plus repair facilities, calibration means, and replacement parts. A dark room is also included.

b. Field Test Facilities. It is planned that both captive and free rocket tests will be made in connection with this program at the Naval Proving Ground, Dahlgren, Virginia. The plans call for a reinforced concrete hood, opening at one end onto a concrete apron 20 ft. wide by 25 ft. long. Imbedded in the concrete apron and in the floor of the hood is a series of parallel rails spaced three ft. on center. The purpose of these rails is to permit the clamping of steel framework and supports for mounting pickups and investigation apparatus. The rails in the hood will anchor the dynamometer stand and absorb the thrust of the rocket. In addition to these main features of the rocket test "pit," facilities are incorporated for drainage and electrical lighting and experimental circuits. It is planned to park the mobile blast research laboratory beside the concrete hood where it can serve as a control center of operation. If the first stand, now under construction, is successful, a total of thirty stands will be built.

c. Naval Gun Factory. Excellent shops and laboratories are available for the design, development, and construction of practically all types of instrumentation including optical, electronic, temperature, pressure, etc.

K. Naval Air Missiles Test Center, Pt. Mugu, California

1. PERSONNEL. The rocket laboratory at Pt. Mugu, California, employs eight engineers and twenty-five mechanics.

2. FACILITIES. Three rocket motor test stands are presently available. Two are set up for the "Gorgon" and "Lark" missiles motor testing and one is planned for testing German equipment. The stands are installed on three concrete anti-aircraft gun emplacements, and control lines are run underground to three heavily reinforced concrete block houses. Only two of the block houses are equipped for testing, and instrumentation consists of standard temperature and pressure recording apparatus, along with additional pressure and temperature gauges and the necessary electrical controls. The surrounding area is protected from the test stands by a 10-ft. high steel piling wall, open on the seaward side in the direction of firing. Each stand is also separated by a short wall in the immediate area of the test equipment.

A large 40 ft. x 100 ft. Quonset Hut immediately adjacent to the test stands is being set up as an assembly shop and storage building. One end is subdivided into a small office, hydraulic test shop, and pump test room.

An elaborate pump test stand is being built for this group under a contract to the Bone Engineering Company, Glendale, California. This stand will be capable of testing and automatically recording the test data of high speed turbo-driven pumps and will supplement or replace a small capacity stand now in use.

A small well-equipped machine shop is located a short distance from the test pits and is capable of producing and modifying moderate size test motors, pumps, etc. A small chemical laboratory and instrument shop are also available for the necessary services of ordinary testing operations such as propellant analysis and instrument calibration.

The test area is rather ideally situated from the standpoint of noise disturbance (by small or intermediate size motors) to the local citizens, and the Mugu Airstrip is close at hand for flight-test work. During certain portions of the year fog will cause considerable interruption of flight testing.

Plans are underway for a permanent facility at Pt. Mugu, and these include a considerable increase in office, engineering, shop, and testing space or buildings. At the moment all present facilities are considered temporary, and pending decision on funds available, the permanent construction will be undertaken.

L. North American Aviation, Inglewood, California

1. PERSONNEL. The propulsion division of the North American Aerophysics Department contains approxi-

mately fifty engineers and twenty draftsmen and technicians.

2. FACILITIES. The Aerophysics Laboratory engineering activities and some of the test facilities are set up in two locations of the main North American Aviation Plant at Los Angeles Municipal Airport.

Engineering and drafting occupy a room approximately 60 ft. x 300 ft., and the electronic, hydraulic, and test equipment listed below is housed in laboratory space of equivalent size.

a. A-4 power plant set up for educational purposes.

b. Supersonic wind tunnel with 1¾" x 4¼" working section at M = 3.2.

c. Burner test setup utilizing the plant air supply for solid fuel ram jet and rocket sweat cooling research.

d. Wave study channel for studying water wave-shock wave analogies.

e. Mock-up and experimental machine shop.

f. Chemistry Laboratory.

g. Electronic and servo laboratory.

Three small test stands are located on local company property. One blockhouse controls two of the pits, one of which is presently set up for a 300-lb. thrust liquid oxygen-alcohol motor and the other for a 1,500-lb. acid-aniline motor. In addition to the test stands, a small assembly shop, a propellant storage building, and a hazardous chemical research laboratory are located on the same site. The research laboratory is used for small-scale testing of sweat cooling, and propellant analysis, etc., that are too dangerous for the main laboratory.

The company has recently leased a site approximately one mile square in Ventura County in the Santa Susana mountains (about thirty-five miles Northwest of the NAA plant). It is reasonably well isolated and is intended as a static test site for large motors, complete missiles, and other hazardous work.

The present plans call for the following equipment and laboratories:

a. Two test stands for 3,000-lb. motors.

b. Rocket component test building (10 test bays).

c. Six additional test stands for small-scale rocket, sweat cooling, and ram jet testing.

d. Liquid hydrogen plant (25 liters/hr. capacity).

e. NATIV missile test stand.

f. Two motor test stands for up to 100,000 lbs. thrust.

g. Heat transfer research laboratory.

The location of the site was chosen primarily for its isolation. It is ideal from this standpoint. No flight test facilities are available there, however; all launching will be done at Alamogordo, New Mexico.

The shops of the main plant will be used for most fabrication work. These are typical shops for an aircraft company and are capable of doing all operations for constructing rocket motors except forgings and castings. Flight test facilities (piloted aircraft) are available at the main plant.

M. Naval Ordnance Testing Station, Inyokern, California

1. PERSONNEL. There are 450 engineers or scientists and 1,500 technicians or mechanics employed at NOTS. In addition there are many construction and maintenance people. The exact number of people assigned to liquid rocket work is not known, but it is probably less than 1% of the total listed above.

2. FACILITIES. NOTS is primarily a guided missile launching and test station, and so is mainly concerned with launchers and tracking devices. The equipment in this line is rather extensive. A large new concrete research building is under construction and will probably be occupied in March. This will house the physics and chemistry research laboratories. It will also have a materials testing laboratory for routine testing.

There is a concrete building for static firing of liquid rockets. It has two test cells and has been used for firing nitrogen peroxide-aniline motors having about 300 lbs. thrust. The instrument room is provided with continuous photographic recording of oil-filled diaphragm pressure gauges.

Additional facilities are available for the handling, forming, and testing of large solid propellant rocket booster units.

N. Ohio State University, Columbus, Ohio

1. PERSONNEL. There are about sixty people working on Liquid Hydrogen and related problems.

2. FACILITIES. There are extensive facilities in the chemistry building for carrying on the liquid hydrogen investigations. There is equipment for studying thermodynamic properties, absorption spectra, heat capacities, heat conductivities, PVT relations, and chemical reactivities. There are plants for the preparation of liquid air, liquid hydrogen, and liquid helium. There are also facilities for research at very high temperatures (high frequency induction heating).

In a separate building about a mile from the chemistry building the work on rocket motors is conducted. It is designed to care for all fire and explosion hazards. Rather large quantities of liquid H_2 can be handled. There are testing machines for studying the physical properties of material (tensile, hardness, impact strength, etc.) at temperatures from 20°K. to room temperature. A small-scale rocket motor test stand is available for testing liquid H_2 and O_2 motors up to 1,000 lbs. thrust.

O. Reaction Motors, Inc., Dover, New Jersey

1. Personnel. The company employs three hundred people, of which eighty-five are engineering personnel, varying from draftsmen to several highly trained scientists.

2. Facilities. Reaction Motors is located on a portion of the land belonging to the U. S. Naval Ammunition Depot, Lake Denmark, New Jersey, and the test stand area is located on property of the U. S. Army Picatinny Arsenal.

The present facility consists of an office building of approximately 3,000 sq. ft. of floor space, a cafeteria with additional office space of 3,000 sq. ft., and two modified two-story barracks buildings of about 25,000 sq. ft. area. About 20% of this is devoted to the engineering department and the remainder to experimental shop, production shop, electronics laboratory, chemical laboratory, heat treating laboratory, welding shop, and supply. The shops are equipped to do any type of work involved in rocket and pulse jet construction except foundry work, which is farmed out.

The test stand area is located approximately three-quarters of a mile from the main buildings and consists of eight test stands made of modified Quonset huts and reinforced concrete. Two additional test stands are located about 300 yards from the above stands and are designed to handle large motors. Linde Air Products has installed several large spherical liquid oxygen containers, some of which are for pure storage and others that may be pressurized for operation of a 6,000-lb. thrust motor for four minutes running time. A large bank of 20-ft. long nitrogen cylinders has been installed for pressurization on any or all of the eight small test stands.

The test stands are suitably located from the standpoint of noise. The nearest airport suitable for piloted flight test operations is approximately fifteen miles.

P. James Coolidge Carter, Pasadena, California

Mr. Carter is a consulting engineer in the field of hydraulics and with particular regard to centrifugal pumps for liquid propellant rockets. With the aid of one or two draftsmen, all design work is done by Mr. Carter and all fabrication and construction is subcontracted out. No test facilities are available and all testing must be done by the contracting agency.

Q. Jet Specialties Company (Subsidiary of G. I. Farman Industries), Pasadena, California

The Jet Specialties Company consists primarily of an office and storage facilities for items on hand. All manufacturing is farmed out to qualified agencies capable of producing the necessary items.

R. White Sands Proving Ground, Los Cruces, New Mexico

1. Personnel. The total figures of personnel located at White Sands and Fort Bliss, and particularly those connected with rocket work, are not available. A large portion of the group, however, is made up of German engineers and scientists.

2. Facilities. Present laboratories consist of a mechanical laboratory (900 sq. ft.), chemical laboratory (1,200 sq. ft.), materials test laboratory (1,000 sq. ft.), and a control laboratory (2,000 sq. ft., under construction). In addition, an air flow test stand is under construction, which consists of 400 cu. ft. of air storage capacity at 3,000 p.s.i. and two 3,000-p.s.i. 70 cu.ft./min. diesel compressors. The stand is a blowdown type and a reducing valve is used to regulate the flow. A static test stand is also under construction for proof-testing of A-4 rocket motors. The larger portion of White Sands is devoted to assembly, testing, and launching of A-4 sounding rockets, and adequate shops and launching facilities are available for this work (49).

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ABSTRACT:

The problems of liquid propellant rocket development and the work being directed toward the solution of these problems are discussed. Factors that affect the design of these engines involve thrust required, propellant used, structural materials, cooling systems injection, ignition, life and operation, operation altitude, and fabrication. The components of a pumping system are described. The development of liquid propellant acid and oxygen oxidizers, and of hydrogen peroxide and nitromethane is outlined.

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